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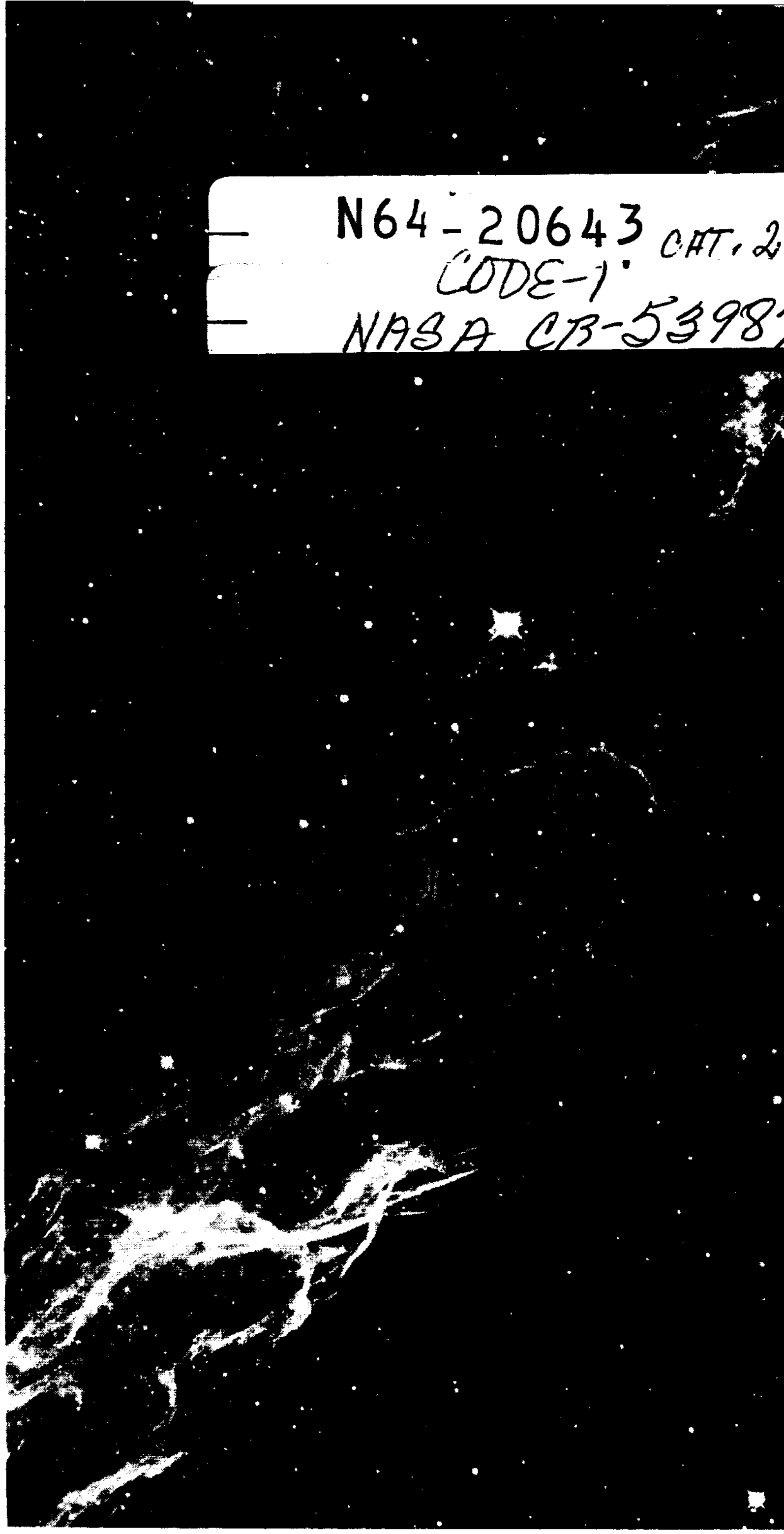

Report No. M-1

SURVEY OF A JOVIAN MISSION

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SURVEY OF A JOVIAN MISSION

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ABSTRACT

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A survey of the requirements for Jovian missions has been made. Scientific questions concerning Jupiter were used to select minimum experimental payloads capable of achieving a significant increase in the detailed knowledge of the planet. Approximate payload weights have been derived from estimates of the scientific and engineering instrumentation, data handling and transmission requirements, electrical power demands, guidance and control requirements, spacecraft structure, shielding, and, when applicable, terminal maneuver energy needed. The use of next-generation launch vehicles has been assumed. Three launch vehicles have been postulated with performance sufficient to cover the range expected in the next decade. The requirements for additional propulsion, and the launch restrictions associated with Jovian missions have been assessed on the basis of these three vehicles.

The addition of a final stage for injection to transfer orbit is shown to be necessary to achieve most missions. The characteristics of a hypothetical high-performance stage are derived for several combinations of mission parameters. Both flyby and orbiter configurations seem "feasible" and further NASA effort appears warranted to improve these estimates to determine more accurately the performance requirements of the additional stage. Author

GLOSSARY

R_J	=	Spacecraft to Jupiter distance measured in Jovian radii
ϕ	=	Half angle of the cone subtended by Jupiter to the spacecraft
θ	=	Equivalent solid angle to 2ϕ
T	=	Mission time
P_T	=	Transmitted power
P_{TH}	=	Receiver threshold
S/N	=	Receiver signal to noise ratio
B	=	Information bandwidth
H	=	Information rate
V_{HL}	=	The launch hyperbolic excess speed
V_{HP}	=	The hyperbolic excess speed at the target
ΔV	=	The ideal velocity, that is, the total velocity increment which must be given to the spacecraft on leaving Earth: $\Delta V = \sqrt{(36,178)^2 + (VHL)^2} + 4000$ Here 36,178 ft/sec is the characteristic velocity for Earth escape launching from Cape Kennedy and 4000 ft/sec is a correction for gravitational and frictional losses during launch
T_F	=	Flight time
δ_a	=	Variation in semi-major axis due to errors in the mid-course correction velocity increment δV
V^2	=	$K\left(\frac{2}{r} - \frac{1}{a}\right)$
V	=	Instantaneous spacecraft velocity

GLOSSARY (Cont'd)

K	=	Gauss' constant for Jupiter
τ	=	Distance from center of Jupiter
a	=	Semi-major axis
D_V	=	The velocity increment required to transfer a spacecraft from its hyperbola into an orbit around a target
ΔV_i	=	Velocity increment from parking orbit
	=	$\sqrt{C_3 + V_e^2} - V_o$
C_3	=	$(V_{HL})^2$
V_e	=	Escape velocity from parking orbit
V_o	=	Earth's orbital velocity
M_{BO}	=	Mass of rocket at burn-out
M_O	=	Initial mass of rocket
u	=	Exhaust velocity
I_{SP}	=	Specific impulse of propellant

SURVEY OF A JOVIAN MISSION

1. INTRODUCTION

The exploration of Jupiter and its neighborhood represents one of several classes of space mission which are beyond current capability and which have not yet been examined in detail. A mission to Jupiter, which orbits at a mean heliocentric distance of 5.2 AU and lies beyond the asteroid belt, typifies in many respects the very deep space mission which will be exposed to difficult but not extreme environmental conditions and operational constraints. The planet is somewhat atypical by virtue of its large mass and corresponding gravitational attraction, however. These general considerations together with the scientific interest in the exploration of Jupiter, as a significant feature of the solar system, are the basis for the present introductory study.

The primary purpose of this survey has been to determine the gross feasibility of the mission and to delineate those factors most critical to its successful accomplishment. Consistent with these goals, a number of estimates, approximations and simplifying assumptions have been made. Every effort has been made to state these restrictions explicitly and to indicate their influence, if any, on the final conclusions.

2. SOME SCIENTIFIC QUESTIONS CONCERNING JUPITER

One of the primary purposes of all planetary exploration is to further our understanding of the origin, evolution, and present state of the planets and ultimately of the solar system as a whole. Study of the giant planet Jupiter can contribute much to our knowledge on these grander issues. While the basic scientific questions concerning Jupiter are treated in a separate ASC document it will be useful to enumerate those which the results of this specific mission study can be expected to illuminate. It is emphasized that the objectives set for this mission study were based only in part upon the relative priorities of the scientific information to be obtained. Other considerations, most notably the state of the art of instrumentation and likely vehicle systems, weighed heavily in the choices.

Among the basic questions attacked are:

2.1 What mechanism supports planetary magnetic fields - e.g. does the dynamo mechanism apply to Jupiter?

2.2 What is the basic aeronomy of a planet other than the earth?

2.3 How are the phenomena associated with radiation belts and solar inputs modified by scaling to Jovian conditions?

2.4 What is the general nature of the giant planet's composition and topography and can this be related to its origin?

3. MISSION OBJECTIVES

The specific objectives of a first mission to Jupiter have been considered based on the foregoing questions, existing astrophysical data and current theories concerning the planet. From these considerations a set of proposed experiments has been compiled to provide the basis for more detailed payload considerations. These experiments form a "basic" package" which is not all inclusive but which, in our opinion, attacks a sufficient number of the primary questions to justify the first mission.

The scientific objectives of this mission are:

3.1 To determine the magnetic field of Jupiter and the magnetic field in the intervening space.

3.2 To determine properties of the Jovian atmosphere, including (a) the temperature, pressure and density of the atmosphere as a function of altitude, (b) the polarization of reflected sunlight, (c) the composition of the atmosphere and (d) the mechanisms that provide the energy balance in the atmosphere.

3.3 To determine the density and energy of the charge particles in the trapped radiation belts.

3.4 To determine the nature or source of prominent observable features such as the famous red spot and the varying colors and the violent motion of the cloud cover.

The following experiments are suggested to aid in resolving the unknown or contradictory features of the observations and speculations.

a. A magnetometer to survey both the Jovian and interplanetary magnetic fields.

b. An infrared spectrophotometer to measure the type and abundance of infrared active molecules that determine the atmospheric energy balance.

c. A microwave radiometer to probe through the atmosphere and if possible measure the temperature of the surface of the planet.

d. A visible and ultraviolet spectrophotometer - polarimeter to measure type and abundance of atmospheric constituents, and provide data concerning total pressure, temperature, aurora, and the red spot.

e. Particle counters to measure flux of cosmic rays, and charged particles either from solar activity or trapped in the planet's magnetic field and micrometeorites.

f. A TV camera to provide photographs of planetary features.

4. BASIC EXPERIMENTS

4.1 Magnetic Field (Magnetometer)

Measurements of the Jovian magnetic field are of paramount interest because the existence and magnitude of the fields are fundamental to the understanding of the planet's magnetic history, the origin of radio emission from Jupiter, and the models describing the planet's interior. The measurement cannot satisfactorily be made from vehicles in the vicinity of the earth, but must be made near Jupiter. The magnetometer system considered will have a dynamic range from 10^{-5} to 100 gauss, so that the interplanetary magnetic field as well as the planet's field can be measured.

There are apparently inconsistent demands of the theories describing the radio emission from Jupiter and the interior of the planet. Although a number of mechanisms have been postulated to describe the source of radio emission, all those that are consistent with observations require the existence of a large magnetic field.(1) The cyclotron theory of radio emission implies a field at the surface of 10^3 gauss, while the synchrotron theory, if correct, reduces the required magnetic field by an order of magnitude.(2, 3) There are also indications that the magnetic field axis is quite asymmetric with respect to the geographical axis.(4) (Plasma oscillations with discharges appear to be ruled out because of frequency and polarization considerations.) It therefore appears that a fairly intense field exists if the proposed explanations are correct.

The existence of this strong magnetic field has interesting implications concerning the models used to describe the interior of the planet. The most acceptable theory concludes that the interior, beginning at a distance of two-tenths of the radius from the surface, is solid metallic hydrogen.[5] This model is based on the known size and mass of the planet, and an assumed equation of state. If the interior is indeed solid metallic hydrogen, it becomes difficult to explain the existence of the magnetic field in terms of a dynamo theory that postulates circulating fluid currents. To avoid this incongruity it has been suggested that the field is a primordial one that has not decayed over the 5×10^9 year lifetime of the solar system. This implies that the temperature of this solid hydrogen has remained less than 800° K for the same time period, because of the dependence of electrical conductivity (and therefore magnetic field decay time) on the temperature.[6] An alternative suggestion is that there exists a core of molten hydrogen metal inside the solid body.[7] This would provide a basis for the operation of a dynamo, but would not explain the asymmetrical field.

The measurement of the magnetic field will not provide an unambiguous conclusion concerning either the interior or the radio emission. However, a measurement of magnitude and orientation of the field will provide a firm basis for further physical interpretation of the planet's properties and, depending on the magnitude, will provide a basis for acceptance or rejection of the radio emission theories.

4.2 Energy Balance (Infrared Spectrophotometer)

The energy balance in the Jovian atmosphere cannot be understood on the basis of the experimental data presently available. The planet receives visible radiation from the sun, absorbs energy, and re-emits lower energy radiation in the infrared, just as the earth and other planets do. The radiation temperature, i. e. the temperature of a planet with the albedo and distance from the sun of Jupiter is calculated to be 86° K.[8] However the measured infrared and radio wave temperatures, believed to be related to thermal emission, are the order of 150° K.[9, 10] The black body radiation for this temperature has a maximum value at a wavelength of 20 μ, and a large percentage of the radiation is between 10μ and 50μ. The only infrared active molecules thus far identified on Jupiter are methane and ammonia, neither of which have any strong absorption bands in this wavelength region.[11] Infrared bands of methane are below 8μ; ammonia has strong absorption bands centered at 10μ and 50μ, but none between these values. Therefore it appears that unless some other mechanism exists for absorption of surface radiation from 10μ to 50μ, the radiation leak would be so great that the temperature would be considerably lower.

The purpose of this experiment would be to examine the causes for the apparent effectiveness of the Jovian atmosphere as a "greenhouse." The instrumentation required is an infrared spectrophotometer for scanning the planet at wavelengths from 10μ to 50μ with a resolution sufficient to detect the existence of other infrared active constituents in the atmosphere. This measurement cannot be made from the earth or earth satellites because of atmospheric absorption and energy considerations. The data obtained would be analyzed to determine the presence and concentration of other infrared active gases. For wavelength regions in which the emission is thermal this experiment will also provide a measurement of atmospheric temperature, and an indication of surface conditions at the polar regions as these are relatively free of clouds.[12]

4.3 Temperature (Microwave Radiometer)

The temperature widely quoted for the planet Jupiter is the 130° K obtained from an infrared measurement in 1926.[9] This temperature is the average of two measurements made on different nights, that yielded 120° K and 140° K, and was made from earth by observing through the 8μ - 14μ atmospheric window. There are more recent radiometer measurements in the microwave region, for wavelengths from about 3 cm to 68 cm.[13] At the shorter wavelengths these results are comparable to the infrared measurements, but as wavelength increases the "temperature" increases to ~ 70,000 °K indicating the radiation is non-thermal. For the infrared temperature there is of course a question concerning the location of the source measured. At this wavelength (8μ to 14μ) it is very unlikely to be the "surface" of the planet, and probably is a measure of ammonia in the atmosphere or clouds. The radio emission measurements show that, in addition to the increase in apparent temperature with wavelength, the emission at the shorter (3 cm) wavelength changes with time which suggests that even at 3 cm there is a non-thermal component to the radio emission.[14]

The instrument proposed for measuring the temperature of Jupiter is a four channel microwave radiometer. The main purpose of this experiment would be to obtain an accurate determination of the planets surface temperature, if indeed a surface can be defined. The four wavelengths should be selected to provide an instrument with the best probability of measuring thermal emission from as low a region in the atmosphere as possible. The wavelengths tentatively selected, 4, 8, 13 and 20 mm, are all below the decameter radiation, which is clearly of non-thermal origin. The 13 mm radiation is near a strong ammonia line, and would give an indication of the ammonia concentration below the clouds. A scan of the planet with this instrument should provide a good measure of surface temperature since the radiation at wavelengths < 3 cm is believed to be primarily thermal.

4.4 Atmospheric Composition and Pressure (Visible and UV Spectrophotometer and Polarimeter)

The Jovian atmosphere, cloud cover, and surface at the polar regions should be examined with a visible and ultraviolet spectrophotometer-polarimeter system, which would yield information concerning the constituents, pressure and possibly the temperature of the atmosphere. The observations suggested are in the wavelength region from 5000A to 1000A, with a resolution of 10A, and sufficient spatial resolution to distinguish limb and terminator effects. The composition of the atmosphere is believed to be largely helium and hydrogen with smaller amounts of methane and ammonia. The methane, ammonia, and hydrogen have been identified spectroscopically, and the helium inferred from its abundance in the solar system.[15] A closer spectroscopic examination of the planet is necessary to identify constituents whose absorption bands are either too weak to be observed on earth, or are below the atmospheric cut-off. A series of closely spaced bands or a continuum from 4200A down to atmospheric cut-off has been observed, the source of which is not yet identified.[16] With the greater intensity and spectral range available to a spectrophotometer on a planetary probe it should be possible to identify this source of emission and other constituents in Jupiter's atmosphere, and permit a more detailed study of the methane and ammonia bands. From the structure of the bands, and by comparison with laboratory measurements, an estimate of the pressure, temperature, and abundance of the gas can be obtained, which will aid in building an atmospheric model.

The identification of other C-N-H compounds in the Jovian atmosphere could ultimately aid in explaining the periodic color changes observed in the clouds. These color changes are variously ascribed to (1) solutions of metallic sodium in ammonia,[17] (2) free radicals produced by reactions with ammonia, methane, and hydrogen,[18] (3) compounds of carbon, nitrogen and hydrogen,[19] and (4) oxides of nitrogen.[20] All of these theories have their adherents and critics, and the question concerning the cloud coloring is still open, as none of these theories appears completely convincing. The measurements proposed permit identification of molecules or free radicals in the atmosphere if they exist in sufficient quantities to cause this coloring, and further, can relate the measurements to the observed color at the time the probe passes.

The same instrument used to scan over various features of the planet will provide additional information. For example a scan over the red spot will indicate the extent to which this region differs spectroscopically from the rest of the planet. A survey of the polar regions will yield information about the atmosphere or surface in the absence of cloud cover. A scan over the limb will yield a pressure measurement, while a scan over the terminator will provide both atmospheric absorption and aurora information. The combination of photometric and polarization measurements should permit an unambiguous fit of a model atmosphere. This is possible because the polarization due to Rayleigh scattering (molecules), Mie (dust particles or thin clouds) and light reflected from the planet can be separated, and a model constructed that fits the measured polarization.[21] The polarization cannot be determined by viewing in one direction or viewing the entire disc, and is not possible from earth because the phase angle is never greater than 11 degrees.

4.5 Trapped Radiation Belts (Particle Counters)

Particle counters should be included to determine both the interplanetary particle flux and to investigate the trapped radiation belts around Jupiter. The existence of trapped radiation belts is implied by the large magnetic fields that were postulated to explain the radio emission. Several models have been proposed to describe these radiation belts, based on an analysis of observed emission.[4, 23] It is suggested that two trapped radiation belts exist, one centered at 1.5 radii and the other between 2 and 3 radii. These belts are pictured as being asymmetrical to the planets geographical axis as a result of an asymmetric dipole moment. This distance is consistent with the fact that radio emission is observed at a distance of 3 radii from the planet. Calculations based on radio emission suggest that there are at least 10^2 particles/cm^3 greater than in the earth's radiation belts, and that a different energy distribution exists.[24] If the estimates of magnetic field strength are correct, it is expected that trapping of high energy particles would be more efficient than in the earth's field. Cosmic ray counters have been included that would provide the mass, energy, and charge of the charged particle flux in interplanetary space. A combination of shielded Geiger counters is considered that will measure the energy and mass of particles in the trapped radiation belts. If the theoretical estimates of the radiation belts are correct, the space probe must be within three Jupiter radii if the trapped radiation is to be measured.

A micrometeorite counter is also included to measure particulate flux in space. Micrometeorite detectors will be included to observe the distribution and size of dust particles throughout the trajectory, in the asteroid belt and in the vicinity of Jupiter. Non-planetary solid particles in space consist of objects ranging in size from several feet in diameter and weighing several tons down to microscopic particles about 0.1 micron diameter and about 10^{-15} grams.

Information about the nature and distribution of particles near the Earth has been obtained from meteor and meteorite studies. Extrapolation of the distribution of matter at large distances ($>$ 1 AU) from the earth is unreliable due to an almost complete lack of data. However meteoroids seem to be concentrated near the earth, and although the reasons for this have not been established, it is not unreasonable to assume that other planets also attract these particles. Whether purely gravitational effects, or meteorite impact on low escape velocity satellites is assumed for the preponderance of dust near the earth, both these mechanisms would be enhanced in the case of Jupiter. However such theories will not indicate the mass flux distribution to be expected. It is hoped that micrometeorite measurements will increase our understanding of these phenomena.

4.6 Photography (TV camera)

The use of photography is clearly desirable, as the resolution of a detailed picture obviously improves as the planet is more closely approached. The problem of taking pictures from a planetary probe is not one of optics but of communication, since a large volume of information is required to transmit one high resolution picture. In fact, this transmitting capability is greater than for all the other experiments combined. The payload weight calculations indicate that both power and weight are not unduly limited, so that the photographic system can be included without compromising experiments. The information to be obtained requires little discussion, as the purpose is simply to provide a much higher resolution and more detailed pictures than could possibly be obtained from the earth. The nature of the targets will include all the visible features of the planet that would be of interest if the photograph were made from earth. For example, of particular interest are the meteorological features, the red spot, cloud movements, and the polar regions where cloud cover is sparse. A great deal of effort has been made to photograph Jupiter through filters, with the purpose of postulating the nature of the red spot.[22] All of these features can be more readily examined with the fly-by or orbiting probe. This instrumentation will also be available for secondary missions. A probe to Jupiter will carry instrumentation close to the asteroids, and pictures from this location could be transmitted before the equipment is needed for the Jupiter experiments.

5. BASIC PAYLOAD

Consideration has been given to the basic subsystems which can make up a payload enabling the scientific objectives to be realized and the data to be processed and transmitted back to Earth. This section outlines the fundamental properties and types of payload subsystems but the selection and configuration of more specific spacecraft is dealt with later in Section 8.

5.1 Magnetometer System

The magnetometer system should have a range from 10^{-5} to 10^{2} gauss in order that the interplanetary magnetic field as well as the near surface Jovian field can be measured. Two types of instrument are suggested which represent the state of the art with regard to measurement and transmission.

5.1.1 Rubidium Vapor Magnetometer

This very sensitive magnetometer measures the Zeeman effect in Rb^{87} vapor in the presence of a magnetic field. The energy difference between the split levels is $h\nu_0$ where ν_0 is the Larmor frequency and equals 6.996 c/s per γ for Rb^{87}. ($1\gamma = 10^{-5}$ gauss). The source of energy in the instrument is a rubidium vapor lamp, the light from which is filtered and polarized and shone through the Rb vapor cell. The variation of the light transmitted through the cell (the light is amplitude modulated at the Larmor frequency) is detected with a silicon light sensitive cell. The application of accurately known bias fields, from three orthogonal coils permits directional measurements and provides a sensitivity down to 1γ. Without undue complication of the data handling system a range from 1γ to 280γ can be covered (frequency range with bias from 60 cps to 2K cps . Measurement of frequency to 0.2 per cent will give a magnetic field accuracy of at least 5 per cent over the whole range. Such an instrument can be constructed within a weight limit of 6 lbs. and a power limit of 5 watts. The environmental precautions necessary are

(a) gas cell temperature limits of $43 \pm 15°$ C and

(b) the susceptibility of the silicon detector to radiation damage.

5.1.2 Rotating Coil Magnetometer

This robust type of magnetometer uses three orthogonal spinning coils to give the direction and magnitude of the magnetic field.

The output is in the form of a sinusoidal voltage proportional to the magnetic field and the speed of rotation. The range of the instrument depends on its design and on the measuring amplifier. Measurements over a range of 4 to 5 decades should be possible and by using a decade selector an accuracy of 1 per cent should be possible over the whole range. The weight will be in the region of 10 lbs. and the total power requirement of the order of a few watts.

5.2 Infrared Spectrometer

Spectrometric measurements are required in the 10μ to 50μ range with as much spectral and spacial resolution as possible. Dispersion can be obtained with a diffraction grating but detection is limited by the wavelength range to a Golay gas cell or a germanium-zinc semiconductor. For a conical view of the planet of half angle $\phi = 1°$, the equivalent solid angle is $\theta = 4\pi \times 10^{-2}$ steradians. If the probe is a distance $(N R_J)$ from the planet (R_J = Jovian radius) then the percentage of surface area viewed is approximately $\frac{N^2 \theta}{2\pi} \times 100\% \simeq 0.02\ N^2\ \%$ per degree. Further using Stefans law the power received at the probe is approximately $\frac{10^{-4}}{N^2}$ watts/cm^2 for a planet temperature of 150° K. It is intended that a spectral resolution of 1μ be provided over the spectrum and that the output be measured to an accuracy of 5 per cent. The total weight of the infrared spectrometer will be about 1 lbs. and its power requirement of the order of 5 watts. The restrictions on the use of a Ge/Zn semiconductor are its sensitivity to radiation and the necessity for operating it at very low temperatures (25° K). The Golay cell although not limited in these ways, has a relatively long time constant for measurement. This refers to state of the art detectors but it can be expected that research in the next few years will produce more suitable solid state detectors.

5.3 Microwave Radiometer

Temperature measurements of the planetary surface should be possible with a microwave radiometer working at wavelengths between 4 mm and 20 mm. It can be arranged to scan in unison with the infrared spectrometer and by using a 30 cm dish a total beam angle of 2° can be used to give more than adequate spatial resolution at distances up to 10 R_J from Jupiter. The input power to the crystal detector should be approximately 10^{-8} watts which will permit measurement to better than 1 per cent. The total weight will be about 7 lbs. and the power required about 1 watt.

5.4 Visible and UV Spectrometry and Polarimetry

These instruments will be used to determine the composition and pressure of the Jovian atmosphere by analysis of the spectrum in the 5000Å to 1000 Å region. A resolution of 10 Å is required over any interesting parts of the spectrum but a system which permits fine and coarse spectral scanning should be incorporated. Dispersion is probably best obtained by a diffraction grating although a lithium fluoride prism is a possibility. Detection can be arranged through an ionization chamber (coarse) or a photosensitive semiconductor crystal. An accuracy of measurement of 5 per cent is required from a signal whose intensity will range over 3 to 4 decades throughout the spectrum. A spatial resolution such that approximately 5 per cent of the planet area can be observed for distances up to 10 R_J should be aimed at, but it is not essential to link the UV scan to the infrared spectrometer. Incorporated in the spectrometer will be a polarimeter which will give adequate information if used only with the coarse detector. The total power requirement will be up to 5 watts and the weight about 10 lbs.

Radiation effects would normally be confined to the semiconductor detector except that at high integrated fluxes there may be coloration of the optical system.

5.5 Particle Counting

Both charged and uncharged particles exist throughout the trajectory between Earth and Jupiter but little data has been obtained beyond the Earth's own orbit. The charged particles to be measured can be classified as cosmic, solar and trapped radiation and the particulates such as micrometeorites and dust.

5.5.1 Cosmic Radiation

Cosmic rays consist mainly of protons of very high energy > 1 Bev. The interplanetary flux is variable with the Sun spot cycle but a mean flux of about 3 $p/cm^2/sec$ has been assumed. The flux decreases inside the atmospheres of planets (typically $10^{-1}/cm^2/sec$ 100 miles above the earth). Cosmic rays, due to their high energy, are easily detected by ionization counters, and arrays of shielded coincidence proportional or geiger counters can be used to give both directional and energy information. The weight of such an array would be about 5 lbs. an d the power requirement only 1/2 watt.

5.5.2 Solar Radiation (Quiet)

Measurements on the solar wind have been performed between Earth and Venus and these should be extended to the region of Jupiter. The energy of protons is less than 3 Kev and of electrons less than 3 eV. (Velocities $\gtrsim$ 400 Km/sec.) The solar wind flux for protons and electrons is about $10^8/cm^2/sec$ which corresponds to a density of about 1 or $2/cm^3$. Measurements on the solar flux can be performed with semiconductor detectors or ionization chambers but magnetic deflection should be utilized to separate electrons from protons and to derive energy data. A simple unit will weigh about 2 lbs. and require less than 1 watt of power but a more comprehensive device may weigh over 10 lbs. and demand perhaps 4 watts.

5.5.3 Solar Radiation (Flare)

During a solar flare not only does the particle flux increase two or three decades but the energy increases to about 800 mev for protons and 1 mev for electrons. Although only a few solar flares may occur during a mission, measurement of flare particles is scientifically significant. An integrating ionization chamber to be used in conjunction with the cosmic ray detectors would be adequate instrumentation at a cost of only 2 lbs. in weight and 1/2 watt of power.

5.5.4 Trapped Radiation

The range of instruments already listed will measure trapped radiation levels which for the earth consist of protons (mainly in 1 mev region) and electrons ($\simeq$ 10 Kev - 100 Kev) at fluxes up to $10^8/cm^2/sec$. The earth belts could be monitored after ejection from the parking orbit as the first operation of the particle counters which would then remain operative throughout the flight. Trapped radiation in the vicinity of Jupiter is postulated about 10^2 times as intense as for the earth and to extend out to about $3 R_J$.

5.5.5 Micrometeorites and Dust

The micrometeorite density will not be constant throughout the mission to Jupiter. In the near earth vicinity impacts occur at the rate of about 7 per hour and vary inversely with distance from the earth to a 1.4 power. However, the results from Mariner II indicate that the region between Earth and Venus has a density some 10^4 times lower than near Earth. It would be useful to compare the near Earth and near Jupiter dust environments to see if some relationship exists which will indicate a source of the dust and a retention mechanism.

A number of dust particle monitors exist at present including large area microphones, crystal flash detectors and pressure cell devices all of which could be used to cover the anticipated range of mass and energy. A weight investment of 2 or 3 lbs. will be required but only about 1 watt of power will be consumed.

5.6 Photography

It is probable that visual pictures from the probe will be restricted to television since normal photographic plates would probably become fogged in the radiation environment en route to and around Jupiter. A television system using say 200 lines could be used to observe either the complete planet or, if sufficiently close, particular areas of interest such as the red spot. The weight of a TV system would be about 20 lbs. and the power requirement about 20 watts.

5.7 Radar Altimeter

Some measure of the distance of the spacecraft from the planet is essential in order to make magnetic and radiometric measurements meaningful. This is best done by a radar altimeter, which is both accurate and rugged. The weight of the instrument lies mainly in the antenna, which need not be very large physically, since a 2 ft. dish can give gains of the order 30 db in the X band. The peak power of such a system would be between 100 W and 500 W assuming a 30 db radar antenna for measurements up to 10 R_J, but the average power consumption would only be a few watts.

5.8 Power Supplies

Electrical power will be required for the spacecraft throughout the whole mission but the power demand will increase with distance from the earth and particularly in the vicinity of Jupiter where all the experiments will be working and where the transmission distance is greater than 4 AU. In the region of Jupiter a power requirement greater than 1 kw seems probable and makes either solar energy (5-1/2 m watt/cm^2) or stored energy impracticable. The most probable power sources are a reactor system or a radioactive isotope source.

5.8.1 Nuclear Reactor Power Supplies

Early in 1955 the Atomic Energy Commission started to develop Systems for Nuclear Auxiliary Power (SNAP). One of these systems, SNAP 8, should be available by 1967 and will generate a thermal power of 600 kw with an electrical output of 30-50 kw. The system which includes the electrical turbine will weigh at least 3500 lbs (current estimate) but this does not include external shielding to protect the spacecraft from nuclear radiation. It will have a thermal radiator of area 1800 square feet and core dimensions of about 15 in. dia x 19 in. long. The design reliability target for the system is 10,000 hrs of continuous operation, although there will be a facility for starting it remotely.

5.8.2 Radioactive Isotope Source

Isotopic sources of power are now becoming available and offer attractive specific powers of up to 140 thermal watts/gm (Po^{210}.) However, for the long half life isotopes most suitable for Jupiter missions it will take years to accumulate kilowatt quantities. The electrical power capability anticipated by 1970 is 1.2 kw for Pu^{238}, 4.7 kw for Cm^{244}, 6.2 kw for Sr^{90} and 4.1 kw for Cs^{137}. Plutonium is an alpha emitter and will need very little shielding but the two beta emitters, strontium and cesium, and curium which emits alphas and neutrons will require shielding. The weight of isotopic power supplies is invested largely in the energy converters which will account for about 1 lb/watt.

5.9 Shielding

A spacecraft traveling more than 4 AU to Jupiter for a period of some 300 days or more will be exposed to many forms of potentially damaging radiation and bombardment. Shielding of the spacecraft must necessarily be a compromise between its weight, its resistance to the various types of penetration, and damage to the payload.

5.9.1 Meteoroid Shielding

Meteoroids are particles of mass generally larger than 10^{-15} gms, of various densities, traveling at velocities between 12 and 80 Km/sec. Such hypervelocity particles will either penetrate a barrier leaving a relatively smooth hole, become embedded in the barrier or just cause erosion and surface deterioration depending on their mass, energy and momentum. To determine the shield required, assumptions must be made about the flux and mass of the meteoroids and the way in which barrier materials react to them.

The meteoroid flux is virtually unknown except in the vicinity of the earth (up to 10^5 Km) where meteor studies have led Beard, Whipple, McCracken and others to define relations between meteor flux and mass. Such a relation is that[25]

$$\phi = K m^{-1.1} \quad (m > 10^{-6} \text{ gms}).$$

The results from Mariner II indicate that between Earth and Venus the flux is 10^{-4} times lower than near Earth but this is based on the very poor statistics of just one impact in 126 days. Further, complete uncertainty exists about the nature of the meteoroid flux in the vicinity of the asteroids and Jupiter.

Although this uncertainty may invalidate any conclusions that are drawn about the amount of meteoroid shielding required for a Jupiter mission, some calculations have been worked out assuming a near Earth flux to prevail for the whole mission to establish the order of shielding that may be required.

A poisson distribution is assumed as defining the frequency of damaging impacts which are all assumed to be perpendicular to the surface. The density of stone (2.8 g/cc) is assumed and the impact velocity is presumed constant at 30 Km/sec. If $P_{(o)}$ is defined as the probability that no penetrations will occur during the mission, then a critical mass m_c can be defined (using the flux mass relation) which has a probability of collision just equal to the probability of no penetration.

Thus $$m_c^{-1.1} = \frac{KAT}{-\ln P_{(o)}}$$

where

T = Time of mission

A = Exposed surface area

K = 1.3×10^{-12} near Earth.

Larger particles than m_c have a small probability of collision and can be ignored whereas smaller particles will not penetrate a shield designed to resist masses of m_c. The degree of penetration produced by meteoroid velocity impacts is by no means agreed, but formulae from two leading investigators are quoted since these bracket most other estimates.

Bjork[26] $$p = 1.09\,(m_c v)^{1/3}$$

Charters and Summers[27] $$p = 2.25\, d_c \left(\frac{\rho_p}{\rho_t}\,\frac{v}{c}\right)^{2/3}$$

p = shield thickness

v = impact velocity

d_c = diameter of meteoroid

ρ_p = density of meteoroid

ρ_t = density of shield

c = speed of light.

It is worth noting that the weight of the shield will be approximately proportional to $(A)^{4/3}$, $(T)^{1/3}$ and $(K)^{1/3}$.

For an area of 100 ft^2 and a mission time of 500 days and assuming $P_{(o)} = 0.9$, Bjork gives p = 0.576 cm and Charters and Summers gives p = 1.218 cm. Using the Charters and Summers figure, the shield weights will be:

Shield	Thickness	Weight
(a) Single aluminum sheet	2p	1380 lbs.
(b) Two aluminum sheets spaced 1"	p/3	460 lbs.
(c) Two aluminum sheets with foam filling	p/10	138 lbs.

Shields (b) and (c) are referred to as bumper shields and offer a considerable saving in shield weight.

5.9.2 Charged Particle and Nuclear Radiation Shielding

Corpuscular and gamma radiation can cause three effects in irradiated components causing displacement, transient or chemical transitions, the latter two resulting primarily from ionization.

(a) Displacement Effects

These are characterized by a long lasting degradation in the performance of a component and are particularly applicable to high purity crystal lattices. The most susceptible components are low frequency silicon transistors and solar cells which are seriously degraded by an integrated neutron flux of 10^{13} n/cm^2. Structural materials such as metals, metal alloys and ceramics can tolerate much higher doses of the order 10^{18} n/cm^2.

(b) Transient Effects

These effects are present usually while the radiation is occurring and for only a short period afterwards and are functions of the irradiating flux rather than the integrated dose. They are manifested in such ways as increased leakage in insulators, noise in solid state circuits and as fluorescence. In general however these effects must be considered for specific payload configurations and circuits although the relatively low interplanetary fluxes over the majority of a mission should not be very troublesome.

(c) Chemical Effects

Chemical changes, due to association or dissociation, would induce hardness in organic polymers and may change the composition of gas mixtures. Typically teflon can withstand approximately 10^5 rads but polythene about 10^{10} rads.

(d) Equivalence of Radiation Doses

There is not and probably never will be one formula which can be used to correlate radiation damage with all radiations of all energies. However comparison tables have been derived which can be used as a guide. Table 5.1 shows such a table for displacement damage in silicon which represents a worst case since silicon semiconductors are probably the most susceptible component in the payload.

(e) Radiation Shielding

Table 5.2 provides a summary of the radiation to be anticipated on the Jupiter mission with the calculated total dose which accounts for the time spent in each region. Comparison of Tables 5.1 and 5.2 shows that shielding is essential for the reactor, the solar wind and the Jovian radiation belts.

The SNAP 8 reactor produces 600 kw of thermal power and therefore undergoes 2×10^{16} fissions per second. From each fission about 1 fast neutron ($\sim$ 1 Mev) and 10 gammas ($\sim$ 1 Mev) escape from the reactor in all directions but need only be shielded against in the direction of the experimental package. To reduce and thermalize the fast neutron flux a combination of heavy and light materials are required and for the gamma rays, equal masses of most materials are equally effective to a first approximation. Preliminary calculations show that a shield of 1000 lbs. weight will reduce the flux to about 10 $n/cm^2/sec$ and 10 r/hr at a distance of 50 feet from the core i.e. just beyond the thermal radiator.

Charged particle shielding simply resolves itself since it can be seen from Fig. 5.1, in the case of protons, that all but the most energetic ones will be stopped by the meteoroid shield (say 2 g/cm^2) and certainly the very high solar flux is very easily stopped. Cosmic rays of course will not be impeded and it is unrealistic to consider shielding against them with the very low flux involved.

Table 5. 1

APPROXIMATE EQUIVALENCE OF RADIATION CAUSING DISLOCATION IN SILICON

Radiation	Energy	Relative Effect	Max Integrated Dose (Si)
Proton	10 Mev	1	$2.5 \times 10^{12}/cm^2$
Proton	100 Mev	0. 3	$7.5 \times 10^{12}/cm^2$
Proton	800 Mev	0. 5	$5 \times 10^{12}/cm^2$
Electron	300 Kev	2.5×10^{-4}	$10^{16}/cm^2$
Electron	1 Mev	3×10^{-3}	$7.5 \times 10^{14}/cm^2$
Electron	5 Mev	10^{-2}	$2.5 \times 10^{14}/cm^2$
Neutron	$\sim$1 Mev	0. 25	$10^{13}/cm^2$
Neutron	Thermal	3×10^{-3}	$7.5 \times 10^{14}/cm^2$
Gamma Photon	1. 3 Mev	2.5×10^{-4}	10^{6} rad

Table 5.2

TABLE OF RADIATIONS ANTICIPATED ON JUPITER MISSION (350 days)

Origin	Location	Radiation	Energy	Flux/cm^2/sec	Total Dose
Cosmic	Parking orbit	Proton	$>$ 10 Bev	10^{-1}	$600/cm^2$
Cosmic	Inner V. A.	Proton	$>$ 1.8 Bev	1	$600/cm^2$
Cosmic	Outer V. A.	Proton	$>$ 1 Bev	3	$2 \times 10^4/cm^2$
Cosmic	Trajectory	Proton	$>$ 1 Bev	3	$10^8/cm^2$
Trapped	Inner V. A.	Proton	$>$ 30 Mev	10^3	$6 \times 10^5/cm^2$
Trapped	Inner V. A.	Proton	$\sim$100 Kev	3×10^8	$2 \times 10^{11}/cm^2$
Trapped	Inner V. A.	Electron	0.04-4 Mev	5×10^6	$3 \times 10^9/cm^2$
Trapped	Outer V. A.	Proton	$>$ 60 Mev	10^2	$6 \times 10^5/cm^2$
Trapped	Outer V. A.	Proton	0.1-1 Mev	10^7	$6 \times 10^{10}/cm^2$
Trapped	Outer V. A.	Electron	20-100 Kev	10^8	$6 \times 10^{11}/cm^2$
Trapped	Outer V. A.	Electron	$<$ 5 Mev	10^6	$6 \times 10^9/cm^2$
Trapped	Outer V. A.	Electron	$>$ 5 Mev	10^3	$6 \times 10^6/cm^2$
Solar Wind	Trajectory	Proton	$<$ 3 Kev	10^8	$3 \times 10^{15}/cm^2$
Solar Wind	Trajectory	Electron	$<$ 2 eV	10^9	$3 \times 10^{16}/cm^2$
Solar Flare	Trajectory	Proton	$>$ 1 Mev	10^5	$10^{10}/cm^2/day$
Solar Flare	Trajectory	Proton	$>$ 100 Mev	10^2	$10^7/cm^2/day$
Solar Flare	Trajectory	Electron	$>$ 5 Kev	10^7	$10^{12}/cm^2/day$
Nuclear (shielded)	Trajectory	Neutron	$\sim$ 1 Mev	10	$3 \times 10^8/cm^2$
Nuclear (shielded)	Trajectory	Gamma	$\sim$ 1 Mev	10 r/hr	8×10^4 rad

Table 5.2

TABLE OF RADIATIONS ANTICIPATED ON JUPITER MISSION (350 days)

(Continued)

Origin	Location	Radiation	Energy	Flux/cm^2/sec	Total Dose
Jupiter trapped	1.5 R_J	Proton	$>$ 30 Mev	10^6	10^{11}/cm^2/day
Jupiter trapped	1.5 R_J	Electron	$\sim$ 1 Mev	5×10^9	5×10^{14}/cm^2/day
Jupiter trapped	3 R_J	Proton	1 Mev	10^{10}	10^{15}/cm^2/day
Jupiter trapped	3 R_J	Electron	20-100 Kev	10^{11}	10^{16}/cm^2/day
Jupiter trapped	3 R_J	Electron	$<$5 Mev	10^9	10^{14}/cm^2/day

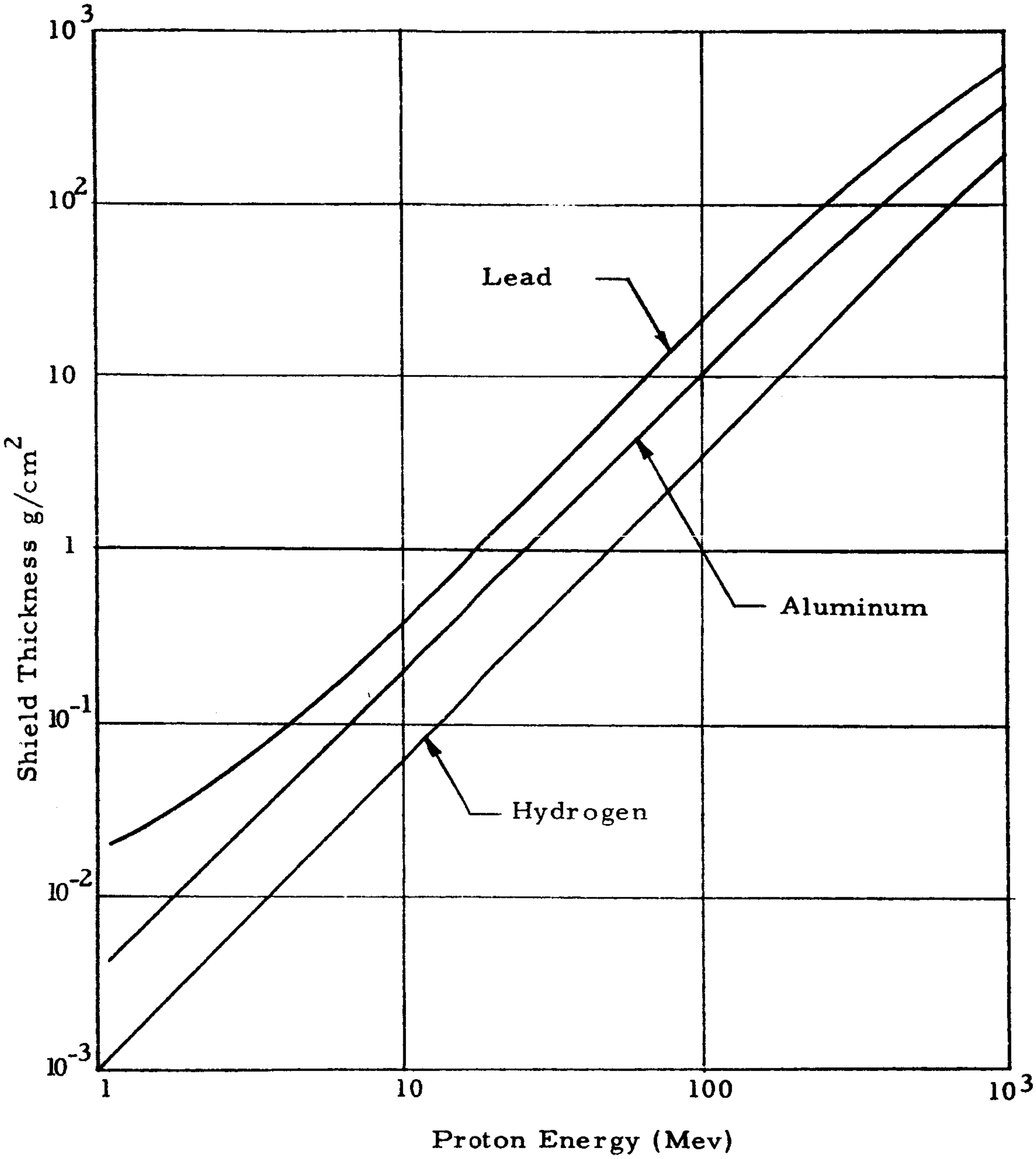

Fig. 5. 1 STOPPING DISTANCE IN GRAMS/CM^2 FOR PROTONS

5.9.3 Magnetic Shielding

This subject is dealt with in Section 6, where a high permeability material is found to be necessary for shielding the experimental package in the vicinity of Jupiter. This probably can be so designed as to act as a meteoroid, charged particle and magnetic shield.

5.10 Central Computer and Sequencer

Under the heading of Central Computer and Sequencer come the following operations

a) Data handling and initial data reduction

b) Multiplexing and format generation

c) Monitor engineering data and generate appropriate commands to control systems

d) Monitor homing system outputs, and order trajectory corrections.

There is little point in discussing details of the system in this report, as, with the exception of 'd' all these functions have been performed on previous space probes.

The possibility of having to employ an on board trajectory correction capability can be estimated once a detailed study of the trajectory and launch vehicle accuracy has been completed. It has been estimated that the extra computation and data storage involved would only add an extra 10^6 bits to the C. C. and S. An estimated total weight of the C. C. and S. is 25 lbs and its power consumption should be less than 5 watts (by extrapolation of Ranger 1 and 2 figures).

5.11 The Data Link

The requirements for a spacecraft data link are defined by

a) the ground support equipment

b) the communication distance

c) the quantity of data to be transmitted

d) the permissable error in transmission of such data.

One assumes that for interplanetary communications, the full D. S. I. F. network would be available, and further, that the proposed modifications will all have been completed by the time a Jupiter mission is flown.

The D. S. I. F. network should, by 1967, have the following capability

Operating frequency (down link)	2300 Mcs
Antenna gain (210 ft diameter)	61 db *
Illumination efficiency	55 per cent
Receiver noise temperature	50° K
Information bandwidth	1 Mcs

Assuming a 6 ft diameter spacecraft antenna, free space losses of L_P = 275 db (at 4. 2 AU), system losses of L_S = 3 db, and a receiver noise temp = 50° K, one calculates a threshhold power P_{TH} = - 110 db m for 1 Mcs bandwidth.

The transmitted power

$$P_T = \frac{S}{N} + 77 \text{ db m}$$

$$= \frac{S}{N} + 47 \text{ db w}$$

Referring to Figure 5. 2, for a bit error probability of 10^{-5} (again, as stipulated for Mariner), the product $\frac{S}{N} \times \frac{B}{H}$ must be approximately 10 db, where H is the bit rate, and B the bandwidth, fixed by D. S. I. F. at 1 Mcs.

For 100 bits per second (b. p. s.) for example, $\frac{B}{H} = 10^4$

* As one goes to larger earth antennas, the noise contribution from Jupiter itself becomes more serious, due to the narrowing of the antenna beam and the consequent increased contribution to the total noise from the Planet. At 2. 3 Kmc, the noise temperature of Jupiter is $\simeq$ 1000° K, and a 60 db antenna on earth would suffer an increase in background noise of 7. 7° K from this.

therefore $\frac{S}{N}$ can be as low as -30 db without increasing the error probability beyond 10^{-5}. Therefore

$$P_T = 17 \text{ db w for 100 b. p. s., i. e. 50 watts.}$$

Therefore, on these assumptions, transmission of digital data can be accomplished for an expenditure of 1/2 watt per bit per second. It must be stressed at this point that this figure is derived from an estimate of the anticipated conditions, and that variations of a db or so in several of the parameters (e. g. antenna gain and system losses) could cause this estimate to be out by a considerable factor.

At present, transmitter efficiencies in the S band lie around 10 per cent. Development of solid state high powered elements could raise this to 20 per cent or more, but for the near future, a bit rate of 100 b. p. s. would require 500 w of D. C. power supplied to the transmitter.

The contributions from the individual experiments to the total quantity of data to be handled are listed in Table 5. 3. It is assumed that the outputs from all experiments will be held in temporary storage prior to the processes of multiplexing and format generation. It can be seen from the table that the main contributors to the bulk of this storage are the U. V. and visual spectrometer, and television.

The bit rate shown is that sufficient to keep the store in dynamic equilibrium (i. e. it never overflows, or empties). This condition of course represents a bare minimum since it assumes a full 24 hr earth coverage, and continuous transmission from the spacecraft which is not necessarily a possibility. The estimate of 30 per cent for engineering data is admittedly somewhat arbitrary, and was obtained by extrapolation of the Mariner format.

Table 5. 3

MISSION INFORMATION RATES

	Accuracy	Measurement Rate	Bits per Reading	Bits per Second
Rubidium Vapor Magnetometer (3 axis)	5%	1 per minute	30	1/2
Rotating Coil Magnetometer (3 axis)	1%	6 plus calibration per minute	30	1/2
Infrared Spectrometer	5%	30 minutes for planetary scan	6000	3-1/3
μ Wave Radiometer	1%	30 minutes for planetary scan	28	1/2
U. V. and Visual Spectrometer	5% intensity 1% wavelength	1 planetary scan per hour	10, 000	3
200 line T. V.	150 km resolution at 3R	1 planetary scan of 20 frames	2×10^6	See text
Particle Counters	Energy 10%	Integrated particle count 3/sec	4	12
Dust Counters		7 /hour	Negligible	
Engineering Data				6

Since the spacecraft will be performing its own attitude control, and terminal velocity correction, the engineering data to be transmitted back might well be less than 30 per cent.

An estimated minimum bit rate is given as 26 b. p. s. (see Table 5. 3), requiring therefore a transmitter of 13 watts and a power consumption of 130 watts. If one assumes an 8 hr transmitting day (e. g. only one D. S. I. F. station used), then these figures rise to approximately 40 watts R. F. and 400 watts D. C., all these estimates being exclusive of T. V.

If the package now includes television, capable of taking one composite picture of the planet (say 20 frames) an additional 2×10^6 bits have to be stored and transmitted. By doubling the bit rate to $\sim$50 b. p. s. this extra information could be transmitted in 24 hours.

In general, the experimental package suggested demands transmitter powers of under 100 watts and consequently the power supply for this item is expected to be under 1 kw. The weight of a 100 watt transmission system will be of the order 8-10 lbs.

5.12 Payload Structure Factor

A structure factor of 10 per cent has been allowed for the payload. Separate allowance has been made for the structure involved in the experimental package, and this is included in the package weight. The choice of 10 per cent is somewhat arbitrary, but is felt to be a reasonable estimate for the 1970's.

6. VELOCITY REQUIREMENTS AND VEHICLE PERFORMANCE

The dynamics of the mission to Jupiter have been considered in three separate parts; launch to earth orbit, transfer from earth orbit to Jupiter and terminal maneuver. This approach has the advantage of simplicity while retaining sufficient accuracy to permit delineation of the major mission features.

6.1 Launch to Earth Orbit

Very rough estimates of the velocity requirements for transfer orbit were coupled with approximate payload weights at an early phase of the study and indicate that worthwhile missions might be accomplished with next-generation chemical rockets if a high performance final stage were added. As a consequence, we have considered as basic launch vehicles, three boosters with an escape capability of 2000-3000 lbs., 6000-7000 lbs., and 60,000-80,000 lbs. These have been designated Vehicle 1, Vehicle 2, and Vehicle 3, respectively.

When computing the velocity increment that must be imparted to the spacecraft one must also consider the energy necessary to escape from earth. The lowest total energy requirements are obtained by considering direct launch from the earth. However, this greatly limits the launch window and we have assumed final injection from a parking orbit. The lower the parking orbit the lower the energy requirement, but low orbits imply higher atmospheric drag and are less stable. We have assumed a nominal 300 nautical mile orbit for this study.

6.2 Transfer to Jupiter

6.2.1 Trajectory Calculations

A basic premise has been the use of impulsively accelerated rockets to inject the spacecraft into a transfer trajectory from earth orbit. It has been assumed that only one gravitating body (the Sun) is acting and that the duration of any thrusting period is sufficiently short that the angular motion around the principle gravitating body is negligible. The flight path then reduces to a conic section and the motion along the path is readily describable. If the time of departure from earth, T_L, and the time of arrival at Jupiter, T_A, are specified, the trajectory is uniquely specified. The ASC/IITRI conic section trajectory system for the IBM 7090[28] was used for trajectory computations.

The launch hyperbolic excess velocity, V_{HL}, required in excess of the asymptotic earth escape velocity is plotted as a function of launch date for several flight times T_F in Figures 6.1, 6.2, and 6.3 The earth orbital period of one year when combined with the Jovian period of eleven years acts to give an approximate synodic period of thirteen months in the excess velocity required. The coplanar Hohmann transfer to 5.2 AU, which gives the minimum velocity requirement, requires a 995 day flight time and a V_{HL} of 8.8 Km/sec. Very long flight times are inconsistent with reliability considerations however, and as a result only relatively short transfers with correspondingly higher velocity requirements have been considered.

The practical difficulties associated with launch schedules make an extended launch window highly desirable. Thirty days has been assumed to be a representative period. Figures 6.4, 6.5 and 6.6 are plots of the required velocity expanded in the region of selected minima. The thirty day launch window allowance adds approximately 1 Km/sec to the minimum velocity requirement. When the departure and arrival time are chosen such that heliocentric central angle is near 180° and the Jovian orbit is off the plane of the ecliptic, a degeneracy occurs in the spacecraft trajectory. The spacecraft orbital plane goes to 90° and a large velocity increment is required for these missions. The peaks seen in Figures 6.4 and 6.6 illustrate this condition.

The hyperbolic excess speed at Jupiter, V_{HP}, is plotted as a function of launch date for various flight times in Figure 6.7. This curve has been derived from the minimum velocity requirements in any launch window. In Figure 6.8 the Earth-Jupiter distance at time of arrival is presented.

6.2.2 Mid-Course Correction

The accuracy of the launch vehicles will probably be insufficient to achieve a reasonable miss-distance at Jupiter without the addition of mid-course correction capability. The desired accuracy for distance of closest approach (measured from the center of the planet) has been specified as $3 \pm 1/2\ R_J$ (based on the requirements of the

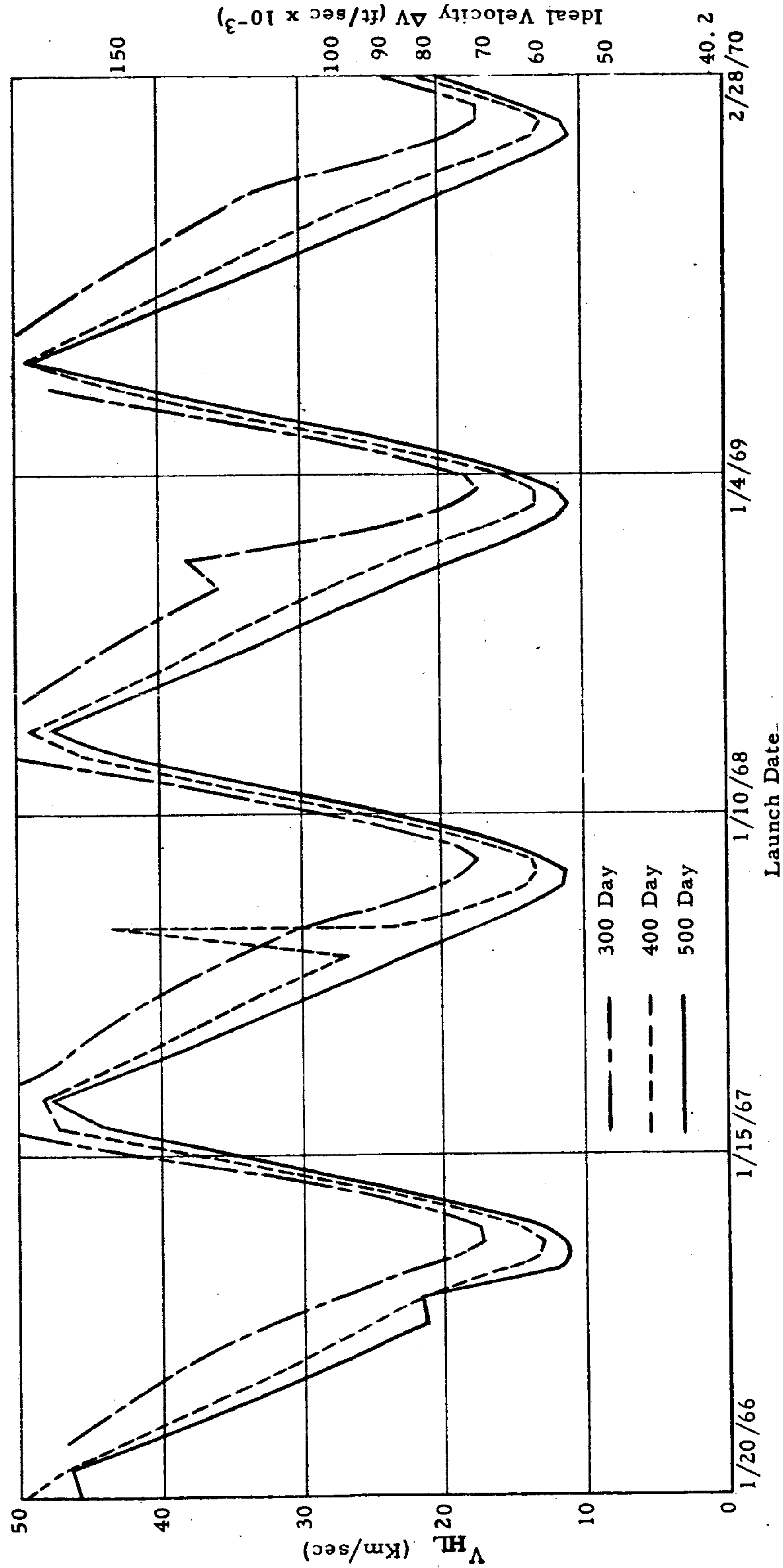

Fig. 6.1 V_{HL} REQUIRED FOR 300, 400, AND 500 DAY FLIGHTS FROM 1966 TO 1970

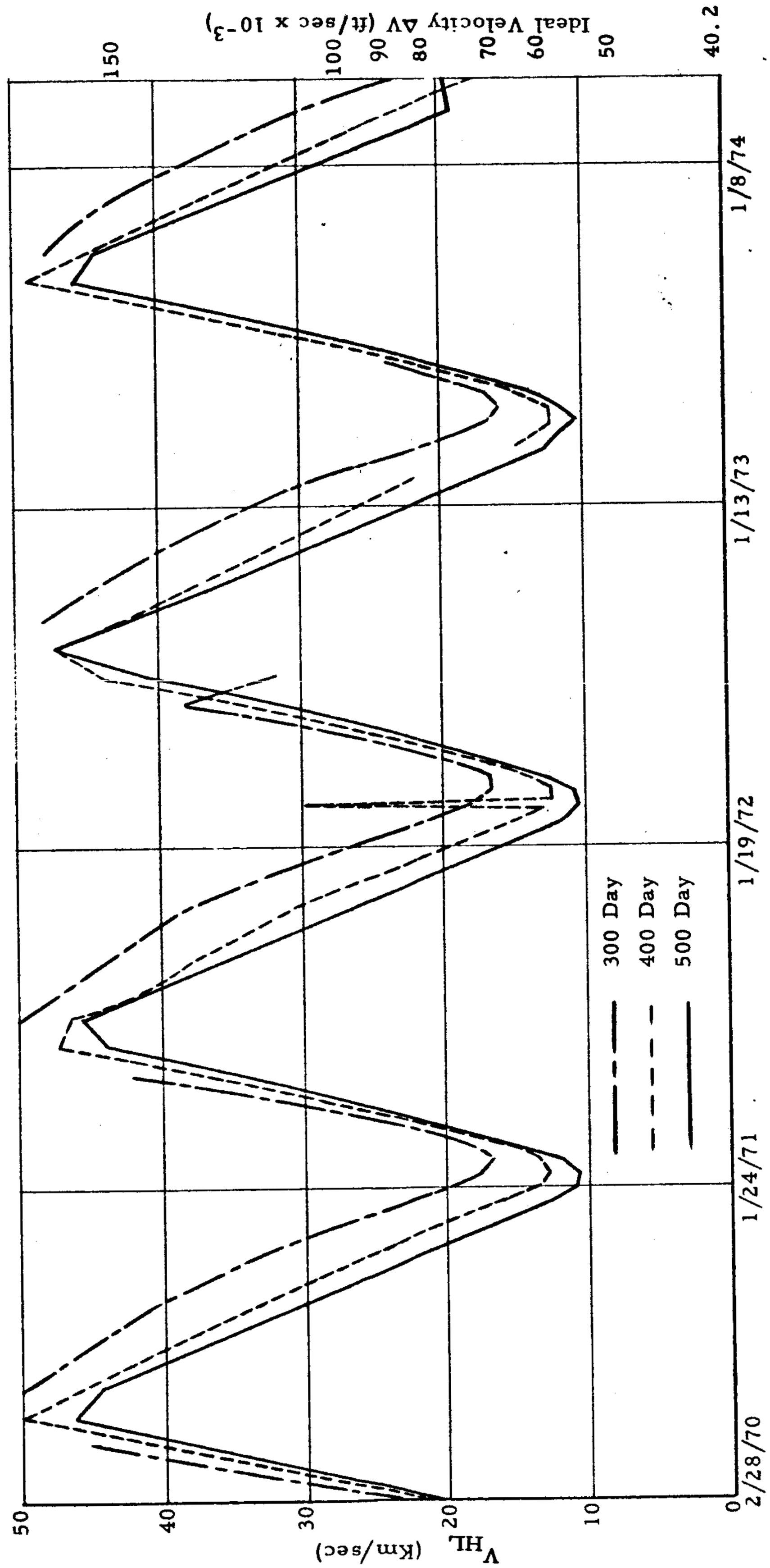

Fig. 6.2 V_{HL} REQUIRED FOR 300, 400, AND 500 DAY FLIGHTS FROM 1970 TO 1974

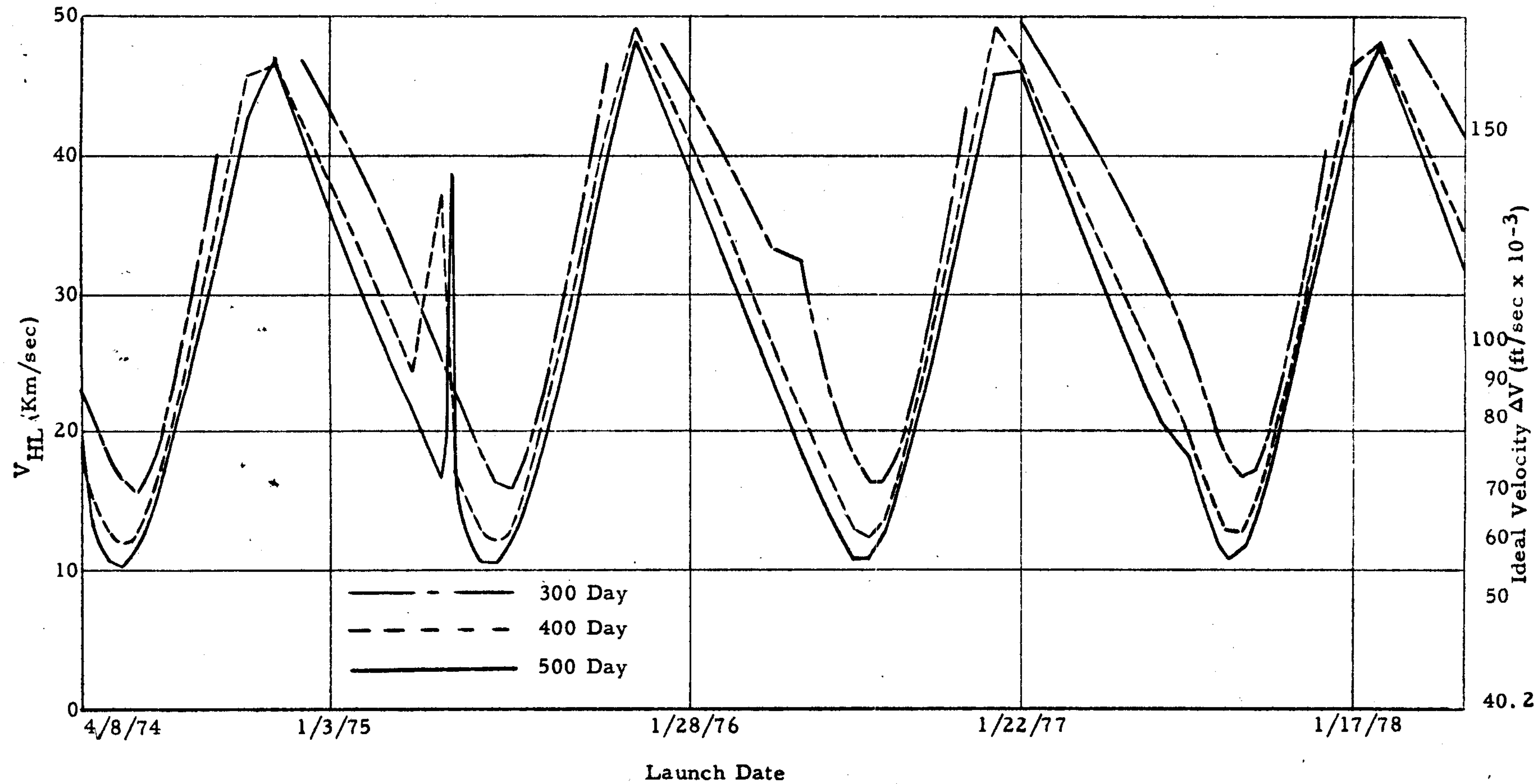

Fig. 6.3 V_{HL} REQUIRED FOR 300, 400 AND 500 DAY FLIGHTS FROM 1974 TO 1978

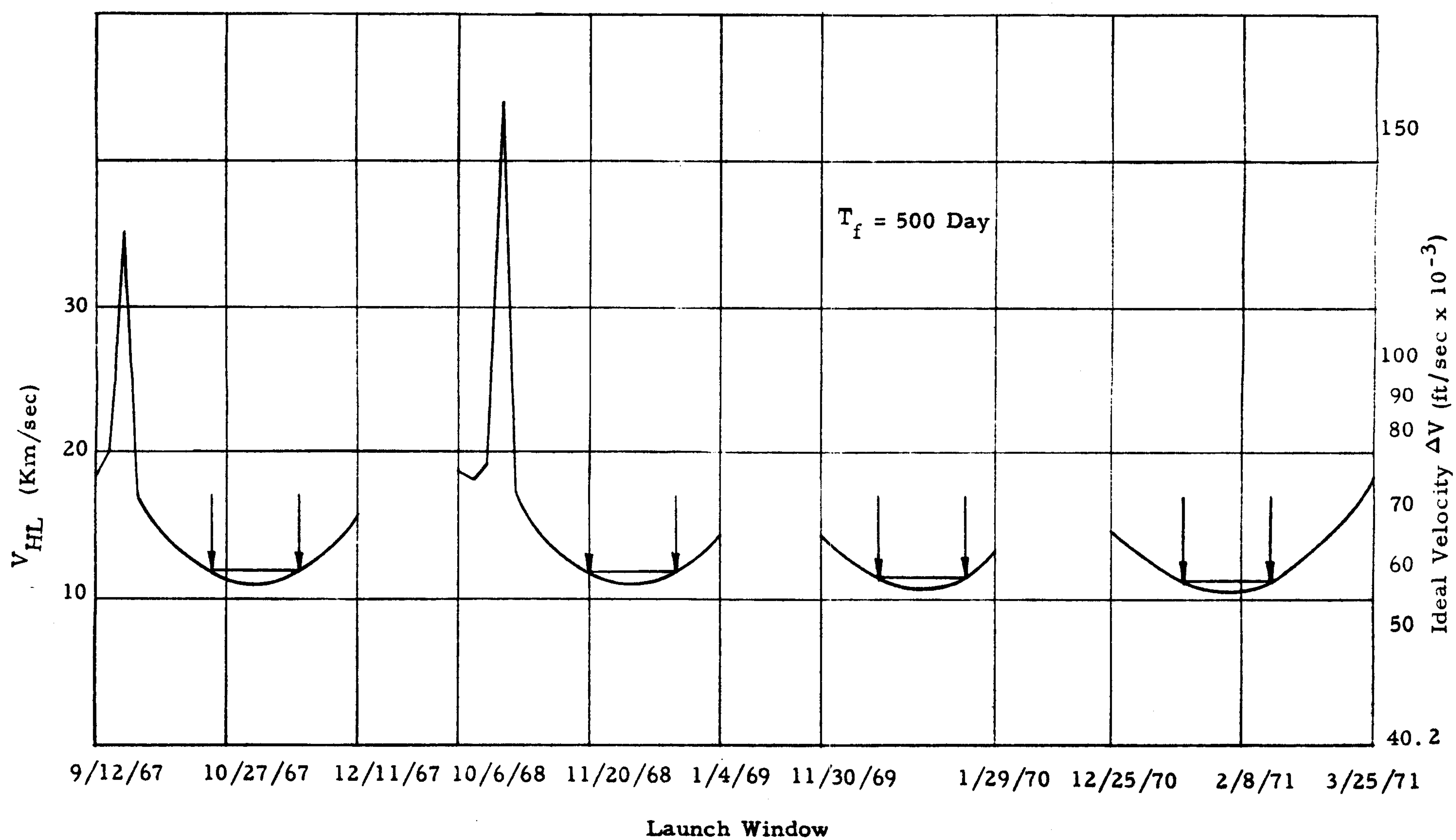

Fig. 6.4 V_{HL} REQUIRED FOR 500 DAY FLIGHT SHOWING 30 DAY LAUNCH WINDOW (1967-1971)

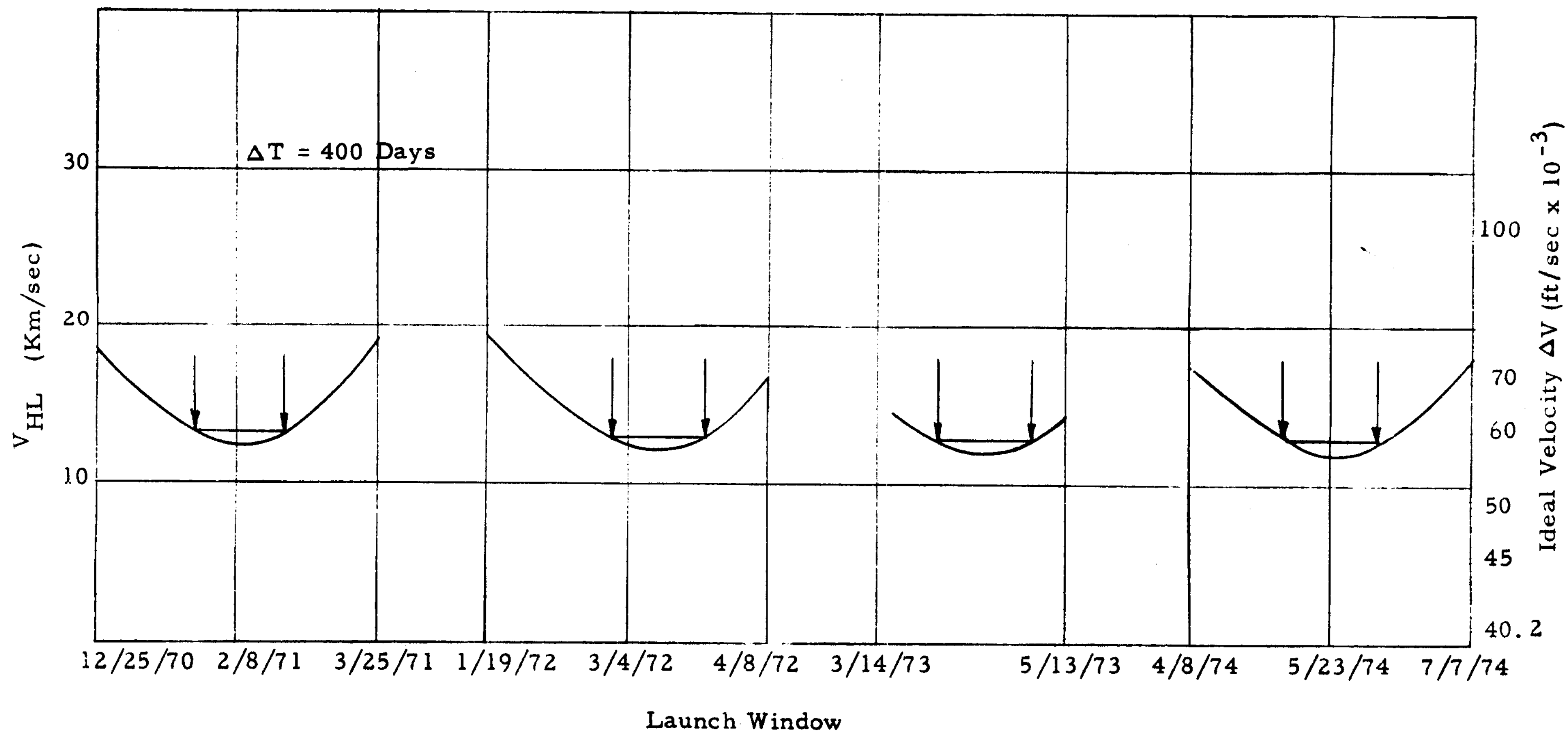

Fig. 6.5 V_{HL} REQUIRED FOR 400 DAY FLIGHT SHOWING 30 DAY LAUNCH WINDOW (1970-1974)

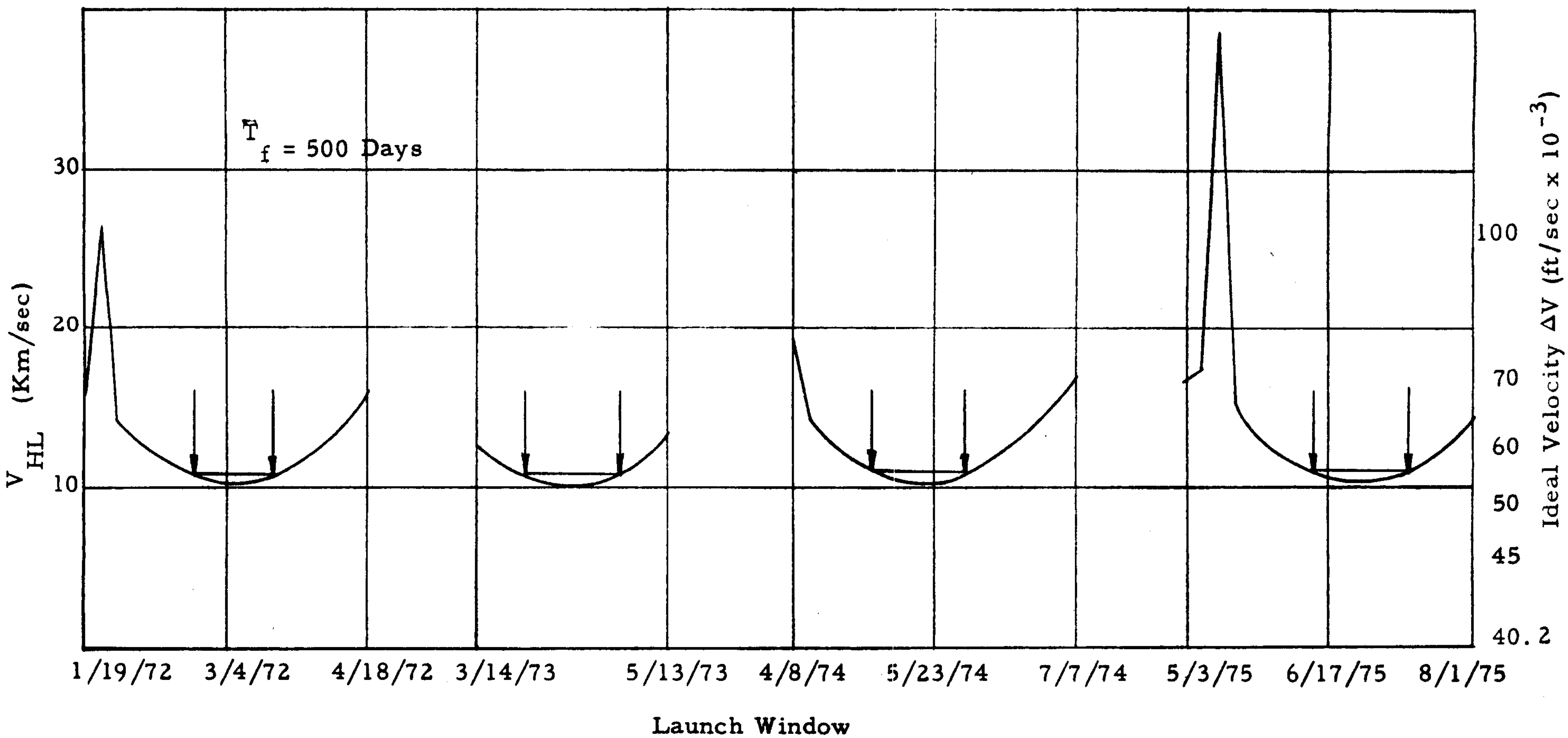

Fig. 6.6 V_{HL} REQUIRED FOR 500 DAY FLIGHT SHOWING 30 DAY LAUNCH WINDOW (1972-1975)

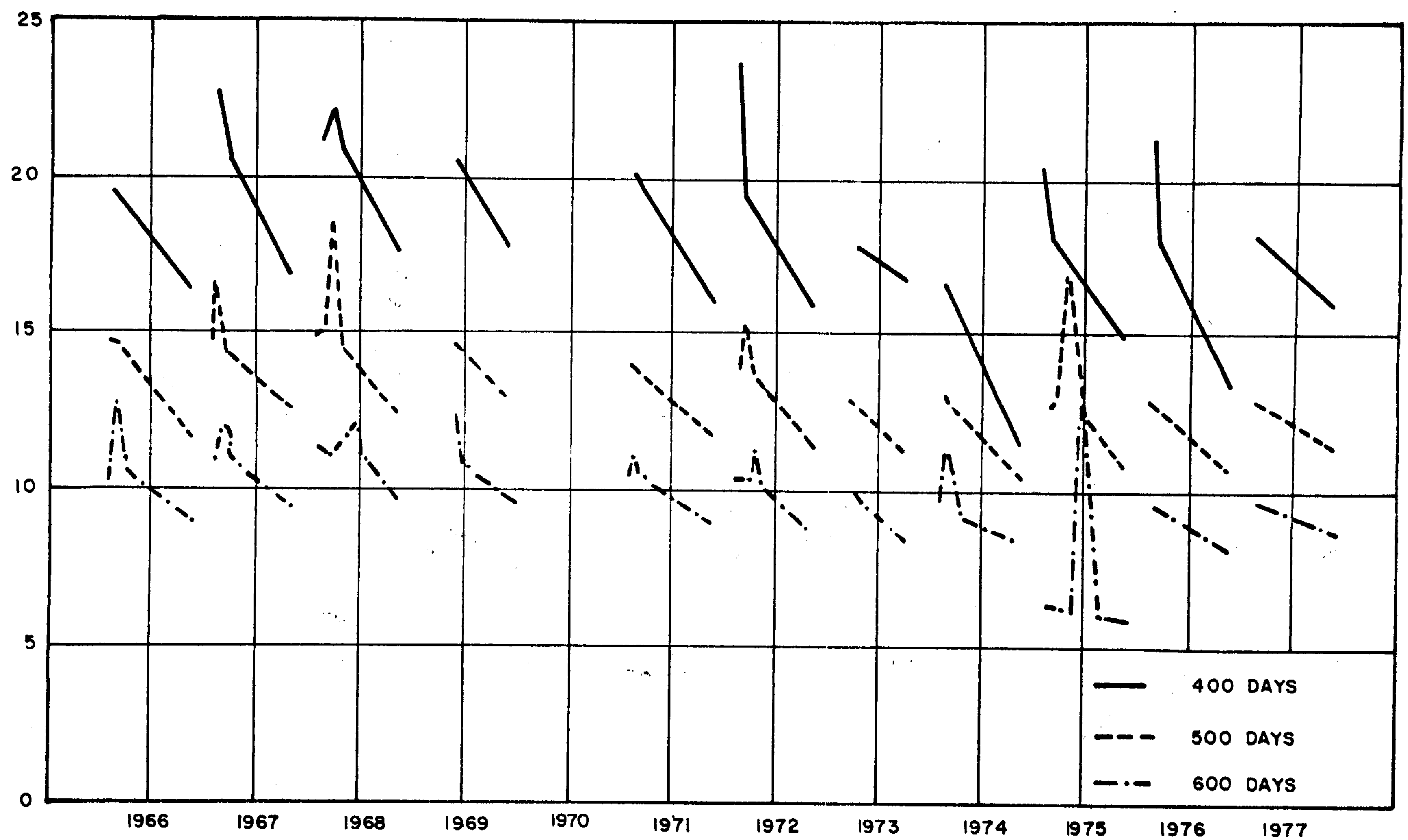

Fig. 6.7 HYPERBOLIC EXCESS SPEED AT JUPITER, V_{HP}, AS A FUNCTION OF LAUNCH DATE AND FLIGHT TIME

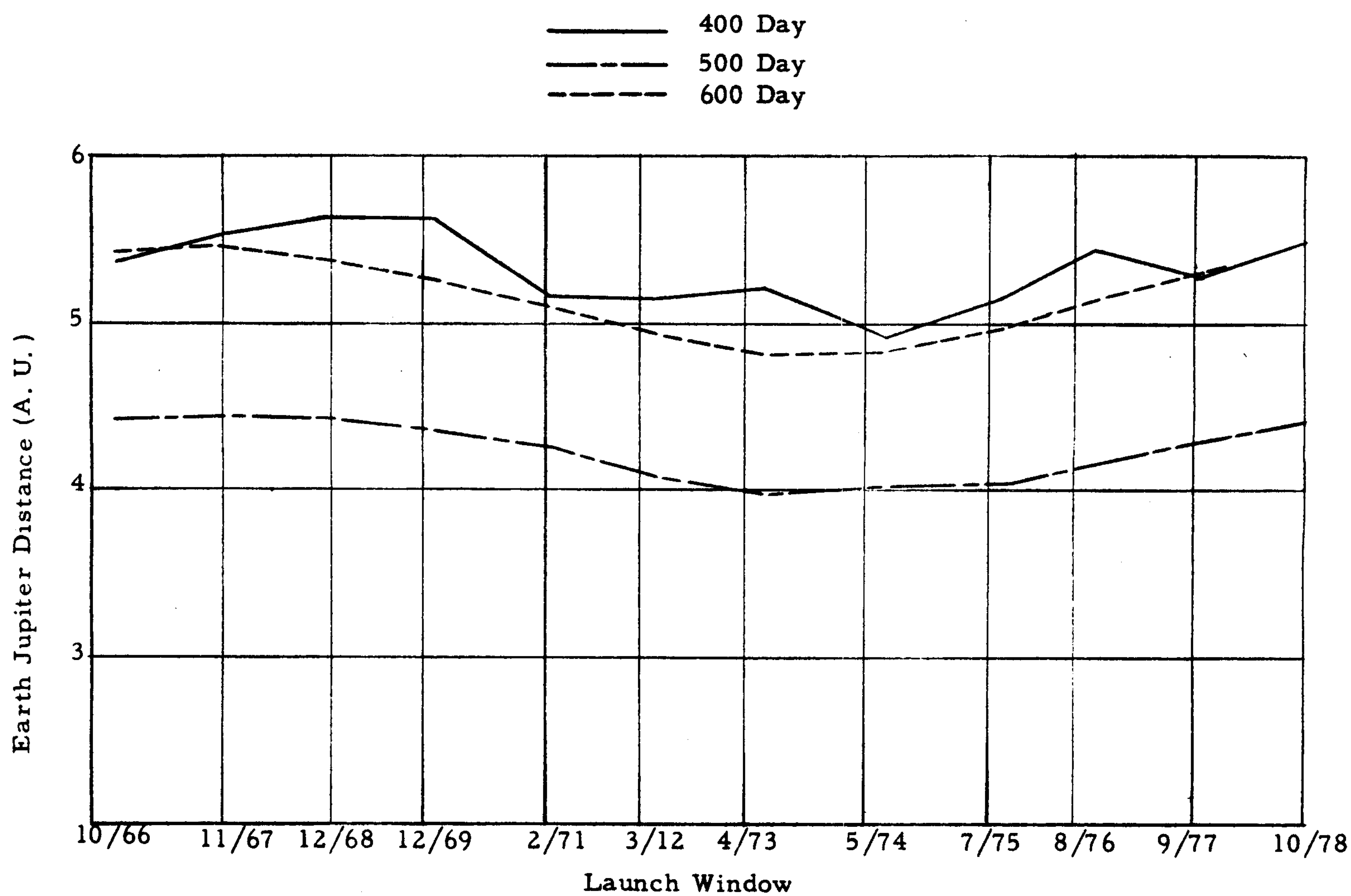

Fig. 6.8 EARTH TO JUPITER DISTANCE AT TIME OF ARRIVAL (COMMUNICATIONS DISTANCE) FOR LAUNCH WINDOWS BETWEEN 1966 AND 1978

scientific experiments. It appears that the first correction to the spacecraft trajectory should be made in the vicinity of the earth, i.e., several days after launch. A mid-course correction system analogous in type to that of the Mariner system has been considered. The Mariner vehicle could accept a commanded velocity correction with a maximum velocity increment capability of 60 m/sec. This velocity increment was adequate to correct for launch vehicle errors and we have assumed that advanced in launch vehicle control will reduce this requirement to 20 meters/sec.

A detailed calculation of the error correction requirement involves determining the error sensitivity of the heliocentric spacecraft orbit, and the accuracy with which the velocity correction can be imposed. Only zero order error sensitivity considerations have been made. We have assumed that the DSIF tracking errors are negligible so that the spacecraft position is accurately known. It is further assumed that the prime source of error after the mid-course correction will be due to variations in cut-off velocity from the mid-course correction engine rather than from the directions of thrusts of this engine. The effect of a variation in cut-off velocity on the semi-major axis of a transfer conic can be obtained from the vis-viva integral given below.

$$V^2 = K\left(\frac{2}{r} - \frac{1}{a}\right)$$

The variation in semi-ma'or axis from an error in velocity is then given by the following equation

$$\Delta a = \frac{2V\Delta V a^2}{K}$$

For typical transfer conditions this error is about $2\,R_J$ for each meter/sec error in cut-off velocity. The product of the semi-major axis cubed and the mean motion squared equals Gauss' constant and the variation in transfer time may be readily inferred. This is of the order of 0.2 days for a 500 day transfer and the above mentioned error in semi-major axis. Since the velocity of Jupiter is about 13 Km/sec, a variation from the desired trajectory to Jupiter of 2×10^5 Km or about $3\,R_J$ would result for a 1 meter/sec error in cut-off velocity.

If the cut-off accuracy is increased to a velocity error of 0.3 meter/sec the resultant error in miss distance is one Jovian radius. This results in an error of $\pm$ 1/2 R_J in the distance of closest approach for a mean value of $3\,R_J$. A cut-off velocity error of 0.3 meters/sec appears to be within the projected state of the art.

If higher accuracies are required for the terminal trajectory then it appears necessary to include a terminal correction capability. This could be achieved by a sensor which would scan the planet Jupiter and determine the variation of the position of the planet with respect to a set of reference axis as a function of time. From this information combined with earth based measurements of the spacecraft position the required terminal correction could be determined. No estimate is given here of the velocity increments required to achieve the increased levels of accuracy.

6.3 Terminal Maneuver

The spacecraft enters the gravitational field of Jupiter with a hyperbolic excess speed V_{HP}. Figure 6.7 gives V_{HP} as a function of launch date for various flight times. Under the influence of the gravitational field the spacecraft enters a hyperbolic orbit given by

$$V^2 = K\left(\frac{2}{r} - \frac{1}{a}\right)$$

where

a is the semi-major axis

r is the distance from the center of Jupiter

K is Gauss' constant for Jupiter

V is the instantaneous velocity.

Substitution of $V = V_{HP}$ and $r = \infty$ yields a. If a miss-distance B (the shortest distance from the center of the planet to the hyperbolic asymptote) is specified, the orbit is defined. The perijove distance for various values of B and V_{HP} are given in Figure 6.9. The orbit corresponding to the suggested fly-by distance is shown in Figure 6.10.

If it is desired to enter an elliptical orbit, the optimum position to thrust is at perijove. A minimum orbit has been selected, again on the basis of experimental requirements. The parameters are a perijove of 3 R_J and an apojove of 100 R_J which corresponds to an orbital period of 45 days. Figure 6.11 is a plot of the velocity increment required to achieve this orbit as a function of the approach speed V_{HP}.

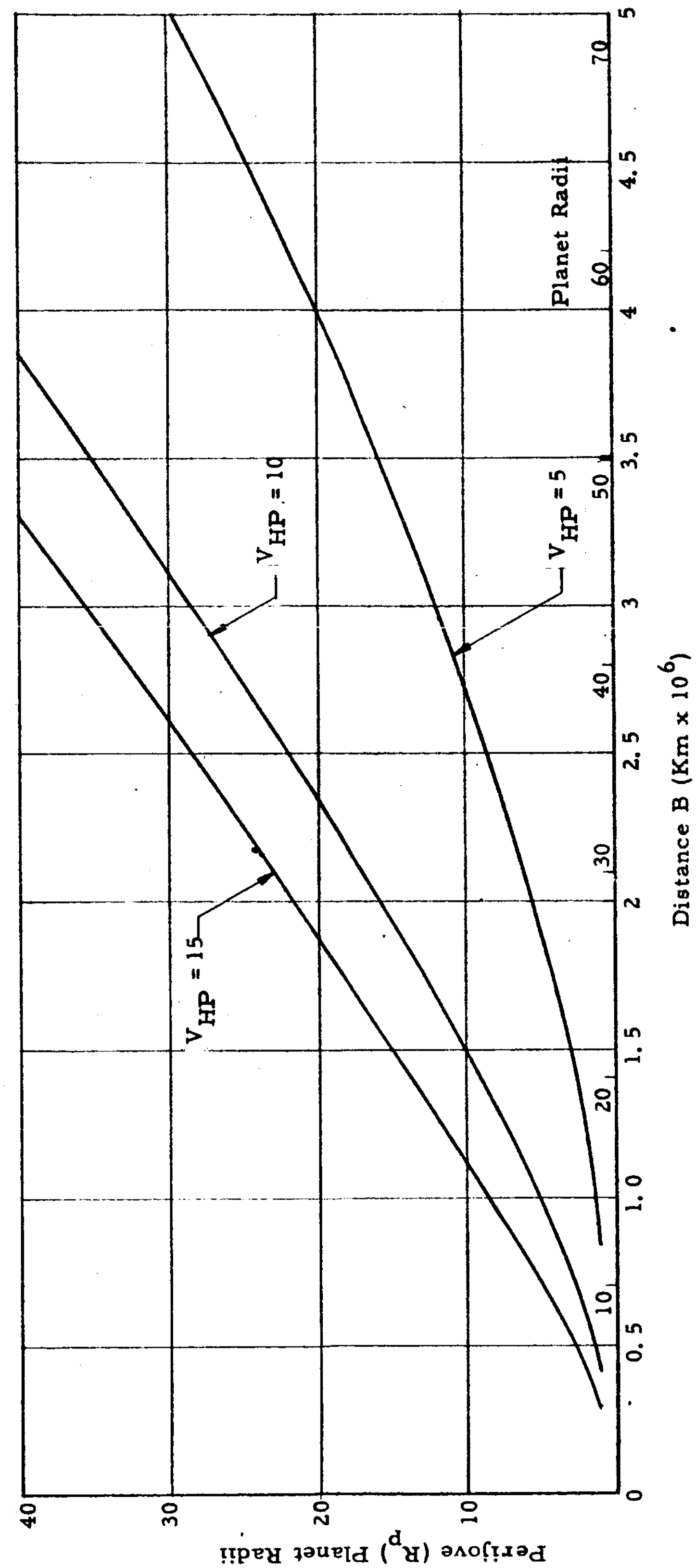

Fig. 6.9 PERIJOVE DISTANCE AS A FUNCTION OF B AND V_{HP}

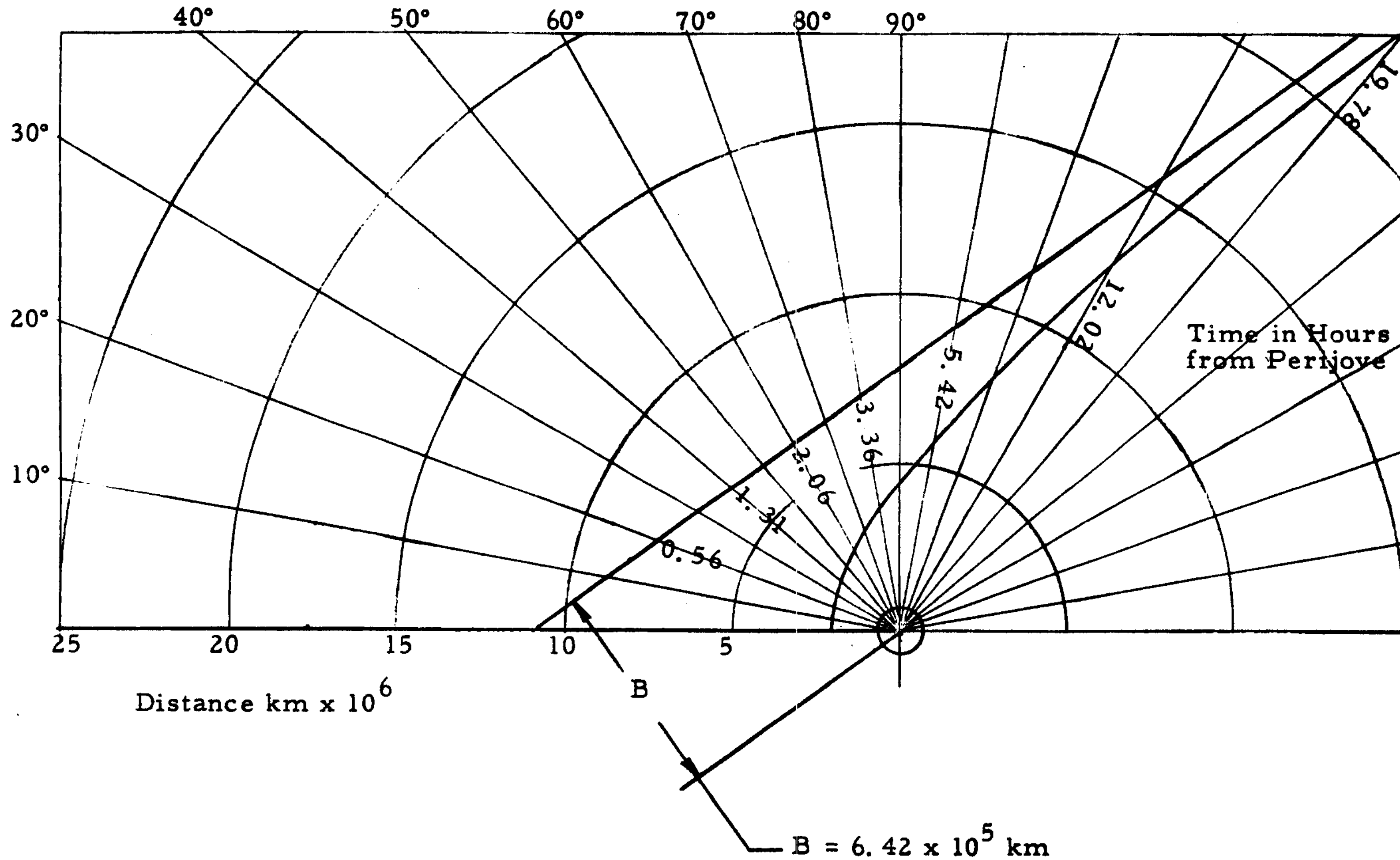

Fig. 6.10 TYPICAL HYPERBOLIC ORBIT FOR JOVIAN FLYBY WITH CLOSEST APPROACH OR 3 JOVIAN RADII AND HYPERBOLIC APPROACH SPEED OF 12 km/sec

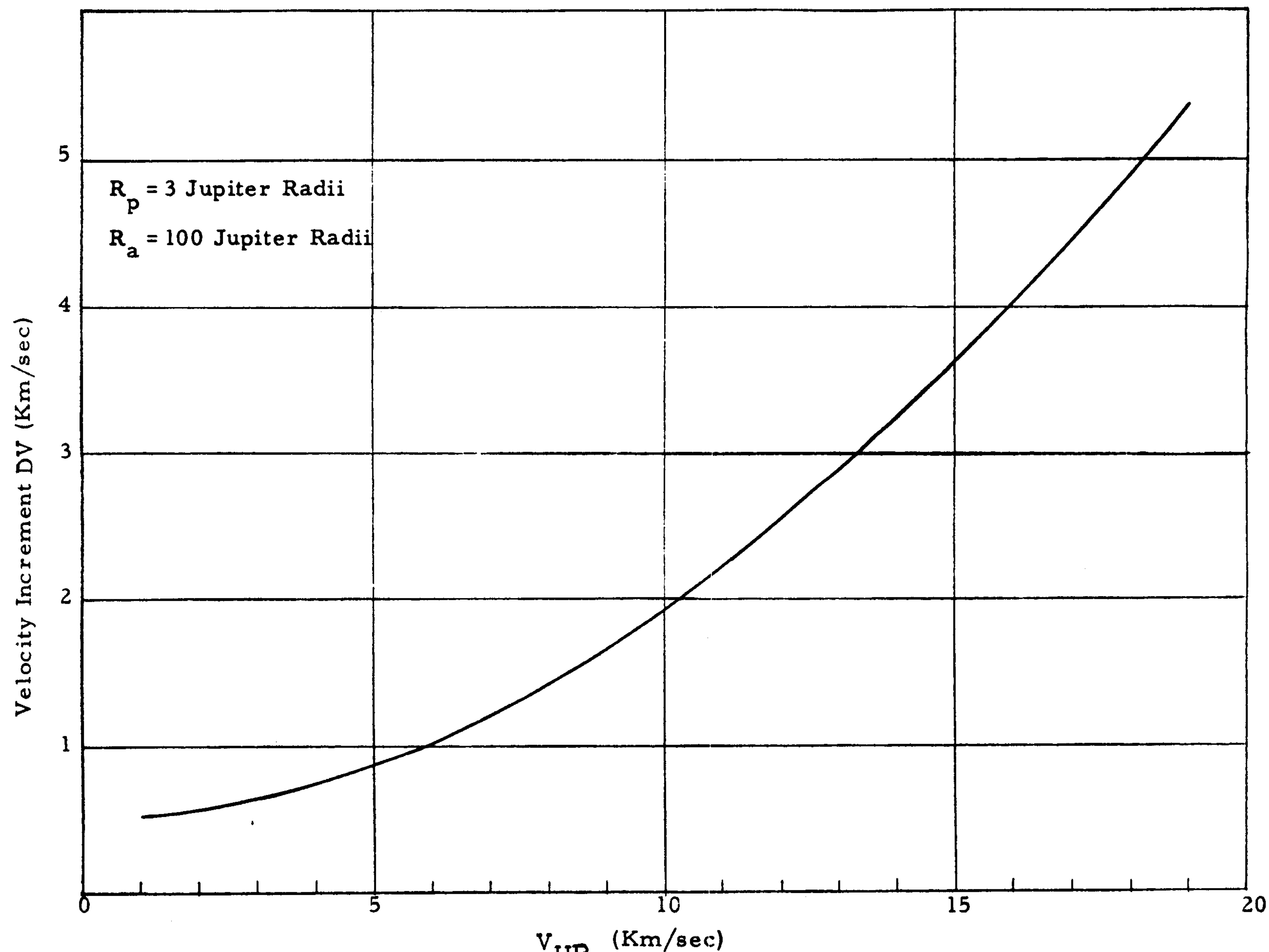

Fig. 6.11 VELOCITY INCREMENT TO ENTER JOVIAN ORBIT AS A FUNCTION OF V_{HP}

6.4 Vehicle Requirements for Injection Correction and Terminal Maneuver

6.4.1 Injection

The velocity requirement for the injection stage from an orbital start is

$$\Delta V_i = \sqrt{C_3 + V_e^2} - V_o ,$$

where V_e is the escape velocity from the orbital position and V_o is the orbital velocity at that point. For circular orbits,

$$V_o = V_e/\sqrt{2}$$

This calculation assumes that the velocity ΔV_i is in the direction of V_o, (i. e. that the injection direction is tangent to and in the plane of the parking orbit). Because of the restraints of the launch site (including range safety) this condition is sometimes impossible to achieve, and dog-leg launch maneuvers are sometimes required. For the Jupiter mission this does not present a problem.

Once the rocket velocity increments ΔV_i, has been calculated one may proceed to calculate the payloads and stage requirements. This is done, for a single stage, through the use of the impulsive rocket equation,

$$\frac{M_{BO}}{M_O} = e^{-\frac{\Delta V_i}{u}} ,$$

where M_{BO} is the mass at burn-out, M_O is the initial mass, and u is the exhaust velocity which may also be written in terms of the specific impulse, I_{SP} (sec), as

$$u = g \, I_{SP} .$$

As previously stated, our starting point has been three postulated next generation boost vehicles. We have furthermore chosen to postulate hypothetical final stages, which are assumed to have just the proper mass to fit the launch vehicle characteristics. No attempt has been made to optimize the staging or the injection orbit.

The final stage characteristics have been chosen to be representative of future LOX - H_2 engines. The I_{SP} (into vacuum) has been taken as 440 sec. We have assumed a payload reduction factor of 10 per cent of the initial mass to account for tankage, structure and mating. The payload reduction factor is applied to the M_{BO} and the net mass at burnout may then be assigned as payload, which is to include experiments, guidance and control, mid-course and terminal (if used) propulsion and supporting structures. These are somewhat optimistic assumptions and the feasibility of the missions to be discussed depends on achieving this performance. We have also used a typical solid propellant stage with an I_{SP} of 225 sec.

The following equation is used

$$P = \frac{M_{PL}}{M_O} = e^{-\frac{\Delta V_i}{g I_{SP}}} - \sigma ,$$

where σ is the overall reduction factor taken here as 0.10.

The following equations may be used to account for small changes in the parameters.

$$d\,P = \frac{\partial P}{\partial I_{SP}}\, d\,I_{SP} + \frac{\partial P}{\partial \Delta V_i}\, d\,V + \frac{\partial P}{\partial \sigma}\, d\,\sigma \, .$$

$$\frac{\partial P}{\partial I_{SP}} = e^{-\frac{\Delta V_i}{g\, I_{SP}}} \quad \frac{\Delta V_i}{g (I_{SP})^2}$$

$$\frac{\partial P}{\Delta V_i} = - \frac{e^{-\frac{\Delta V_i}{g I_{SP}}}}{g I_{SP}} ,$$

$$\frac{\partial P}{\partial \sigma} = - 1 \quad .$$

Three vehicle performance curves are included. Figures 6.12, 6.13 and 6.14 give the total spacecraft weight vs V_{HL} for the three launch vehicles. The calculation of the payload was made using ΔV, which includes the energy required to escape from the earth as well as that to achieve the Jupiter transfer, but the results have been plotted as a function of V_{HL} for convenience. Figure 6.15 shows the spacecraft weight required to place 2000 lbs. in orbit as a function of the terminal velocity increment DV.

6.4.2 Rocket Requirements for Mid-Course and Terminal Corrections

The rocket engine for mid-course and terminal guidance must use a storable fuel that can be restarted or must be divided into parts to be used for the various corrections. We have assumed an I_{SP} of 350 which could refer either to hydrazine or to solid propellants.

The terminal velocity increment for a specific mission can be determined from the data plotted in Figure 6.16. The maximum requirement for an orbit of R_P = 3 Jupiter radii and R_A = 100 Jupiter radii during the next 15 years occur in 1968 when a DV of 3.1 km/sec is necessary to orbit. Note that a nominal extension of the flight time greatly reduces the value of DV.

The total velocity requirement consists of a 0.02 km/sec for mid-course correction and $<$3.1 km/sec for terminal

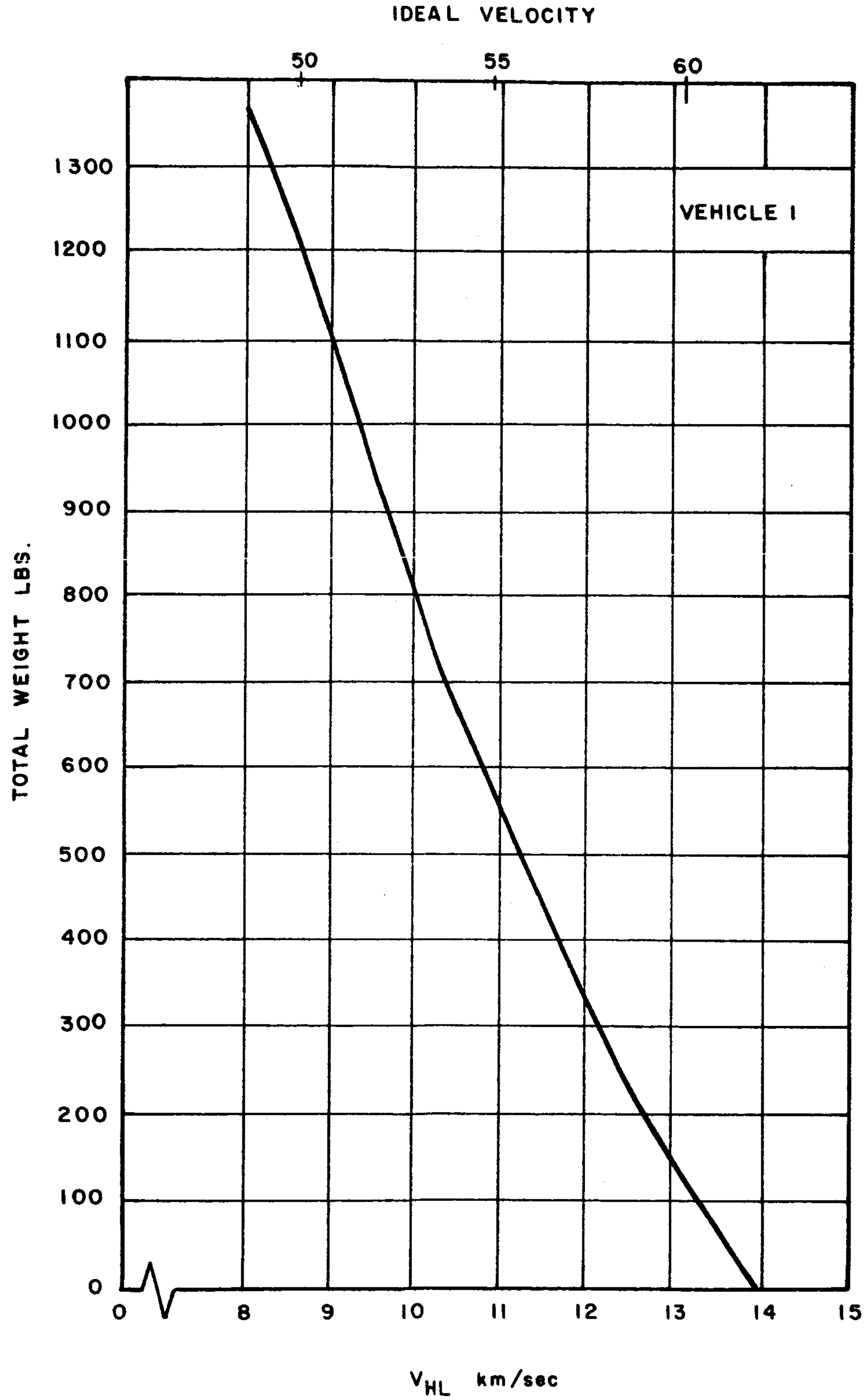

FIG. 6.12 TOTAL SPACECRAFT WEIGHT VS. V_{HL}

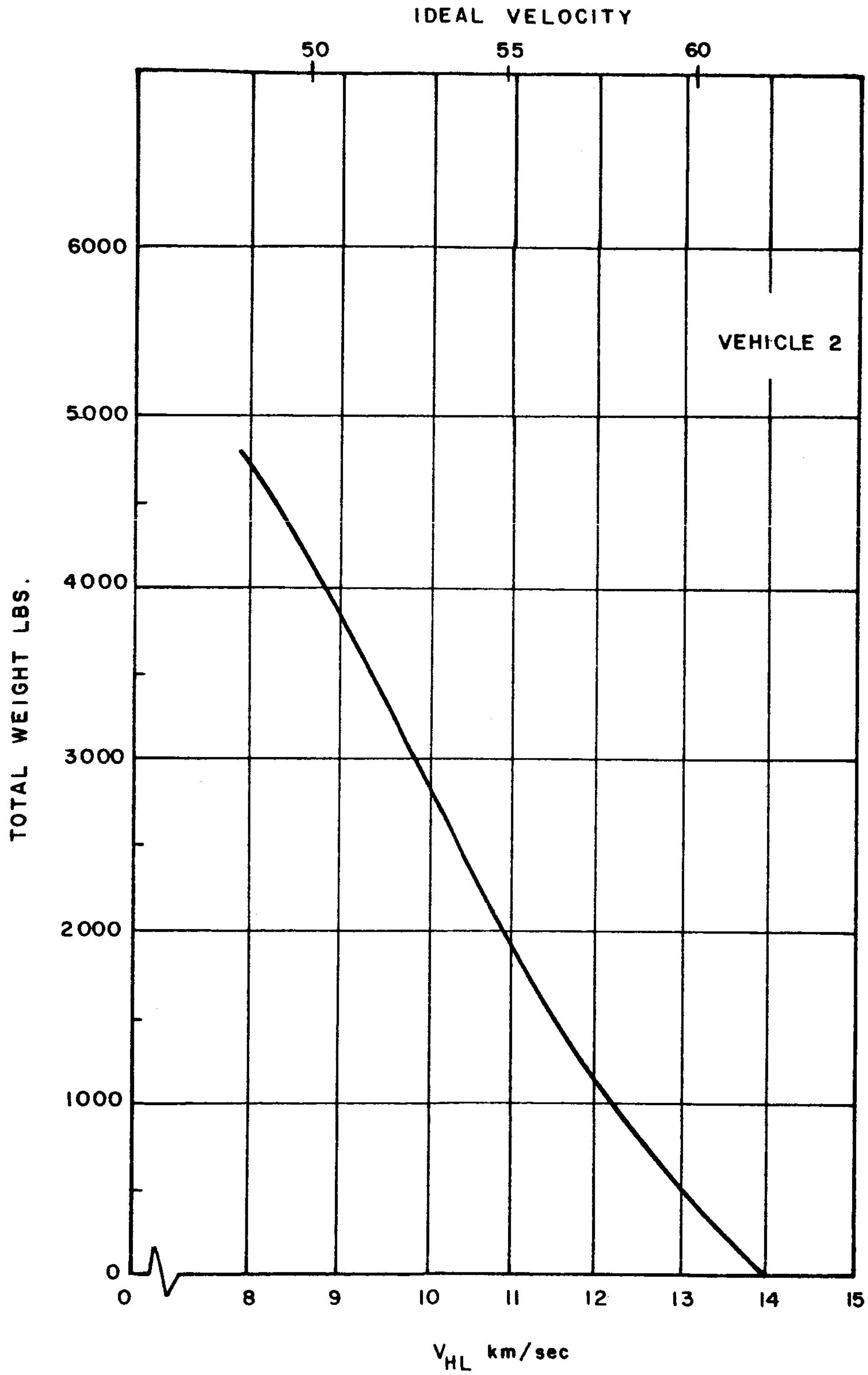

FIG. 6.13 TOTAL SPACECRAFT WEIGHT VS V_{HL}

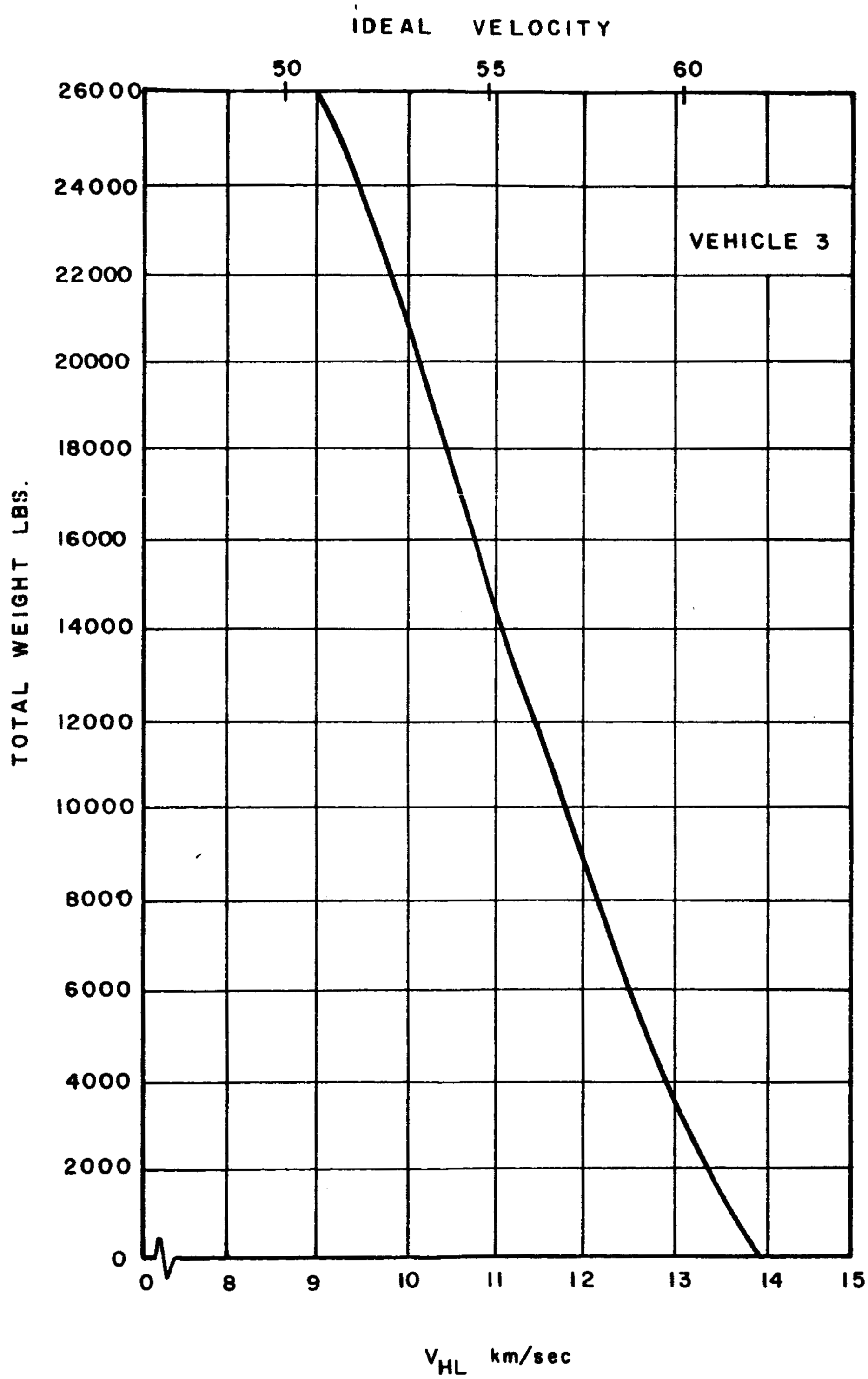

FIG. 6.14 TOTAL SPACECRAFT WEIGHT VS V_{HL}

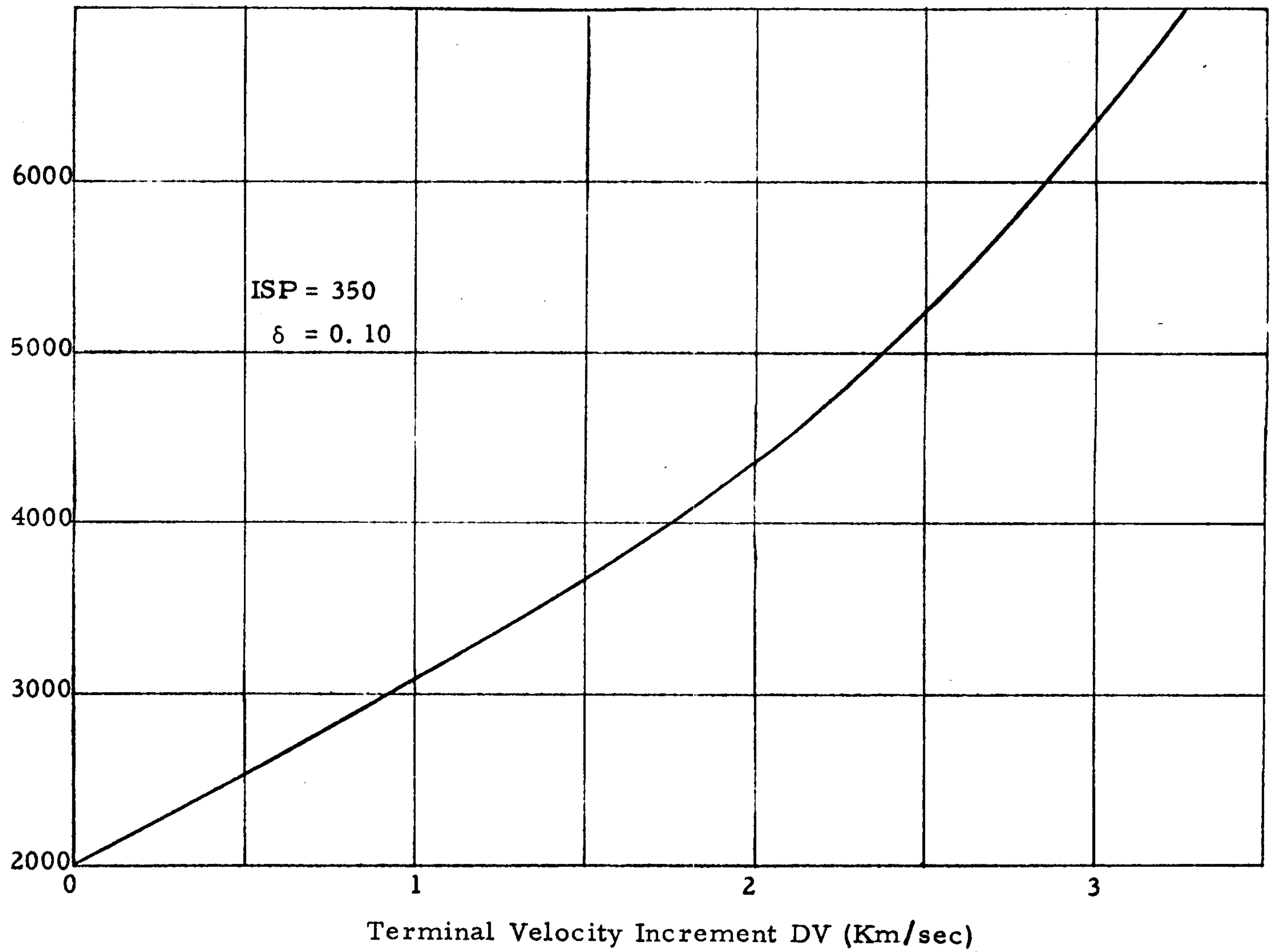

Fig. 6.15 SPACECRAFT WEIGHT REQUIRED TO PLACE 2000 Lbs. IN ORBIT AROUND JUPITER AS A FUNCTION OF TERMINAL VELOCITY INCREMENT

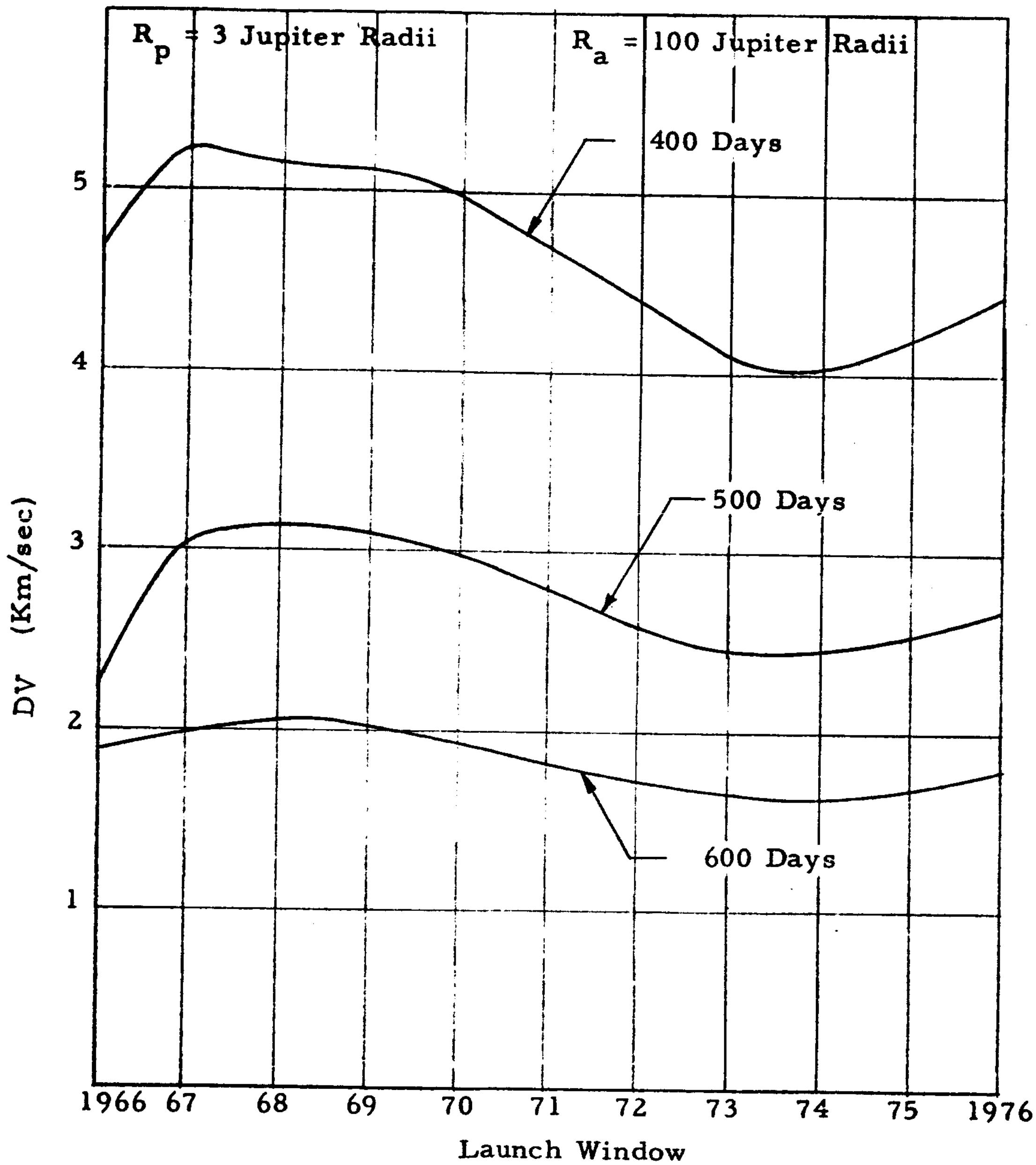

Fig. 6.16 DV REQUIRED TO CHANGE FROM A HYPERBOLIC TO AN ELLIPTICAL ORBIT AS A FUNCTION OF LAUNCH DATE AND FLIGHT TIME

maneuver to the Jupiter orbit desired for the chosen parameters of the transfer orbit. Thus in a payload designed to orbit, the terminal velocity increment swamps the mid-course correction and the scientific payload is limited to approximately 25 per cent of the total vehicle mass (to provide capability of 3 Km/sec).

For a smaller vehicle with no orbiting capability one must provide a small rocket engine to give the required 0.02 Km/sec. For such a small velocity increment the fuel requirements are much smaller than the weight of the hardware and subsidiary equipment. Therefore, a fixed weight allowance based on extrapolations of the Mariner system has been used.

7. MISSION BOUNDARIES

Before discussing any specific mission to Jupiter it is advisable to define the boundaries common to all such missions. These may be divided into classes

a) Environmental constraints
   - Temperature
   - Radiation damage
   - Magnetic field effects
   - Meteorite damage

b) Functional constraints
   - Reliability

c) Experimental constraints
   - Spacecraft stability
   - Elliptical orbit parameters

7.1 Environmental Constraints

The permissable temperature range over which the spacecraft will operate is largely defined by the semiconductor devices on board. The instrument most sensitive to temperature changes is the rubidium vapor magnetometer, but this can easily be provided with its own thermal control. Silicon semiconductors for space at present have a temperature range of 233-333°K. Since the equilibrium temperature at 5 AU is only 140°K some heating of the individual electronic packages is indicated. This subject has not been investigated in detail, but it appears that the surplus heat from the power supply will be more than enough to ensure the correct thermal environment for the solid state equipment, since neither reactors nor isotopic supplies are likely to have efficiencies much greater than 15 per cent or so, leaving 85 per cent of the dissipated energy as heat.

It appears therefore, unless the flight path takes the spacecraft nearer the sun than 0.7 AU (where the equilibrium temperature of a rotating body is 331°K) no elaborate thermal system other than local heating will be required for the instrumentation.

The limitations on Jupiter missions caused by radiation and meteorite damage have been mentioned earlier (Section 5.9), and may be interpreted as imposing a weight penalty which increases with mission time.

The magnetic field in the vicinity of Jupiter has been variously estimated with a generally accepted maximum value of 1000 gauss. At a distance of 3 R_J the field may be expected to be lower, probably still of the order of 100 gauss, however. It is therefore essential to incorporate some measure of magnetic shielding in the spacecraft design, since many parts of the experimental package and telemetry system are sensitive to magnetic fields and would be rendered inoperative by fields of this order (e.g., television system traveling wave tubes, etc.)

Magnetic shielding can be accomplished by providing a cover of high permeability and high magnetic saturation. This cover can either be an overall shield, individual shields, or a combination of both.

Minimum shield weight will be achieved by the combination shield, but the use of an outer shield leads to the interesting possibility that it could also be used as the meteorite bumper. For complete shielding from a field of 100 G (corresponding to 3 R_J) the shield thickness for a 20' by 4' diameter spacecraft would be approximately 8 mm giving a weight penalty of 250 lbs. This is of the same order as the shielding requirements for protection from meteorites, and is hence an attractive configuration. A combination of radiation, meteorite and magnetic shields could perhaps cut the overall weight penalty by a factor of two or more. The obvious design philosophy is to estimate the extent of meteorite shielding required, and if this is sufficient for the magnetic shield, to add individual shields for each electronic package. If the spacecraft is to approach closer than 3 R_J, this latter condition will certainly prevail. For this reason and because 3 R_J is thought to be the outer limit of the Jovian radiation belts, we have selected 3 R_J as the distance of closest approach.

7.2 Reliability

This area has not been investigated in detail, since to derive a meaningful estimate (insofar as any reliability estimate is meaningful) requires a more detailed scapecraft specification than has been attempted here. However, it seems reasonable to assume that advances in manufacturing techniques, inspection, testing and basic understanding of failure mechanisms over the next decade should permit reliable operation of a complex spacecraft for periods of as long as 500 days. Thus, with the single exception of the telemetry system, no allowance for redundancy has been made in the subsequent payload weight calculations.

7.3 Spacecraft Stability

The stability and orientation of the spacecraft must be assured to some degree in order that all the experimental data can be pieced together. During the cruise mode the major restraint is on the transmission of data. If the Sun or Earth sensors become unlocked it would be sufficient to prevent telemetry during the period of re-orientation.

In the vicinity of Jupiter, the spectrometers and the radiometer will scan the planet and make measurements on a series of predetermined areas. The viewing angle will be chosen for these instruments so that 5 per cent of the planet surface will be seen from a radius of 10 R_J. This is equivalent to a conical half angle of approximately $2\text{-}1/2^o$. An instability which changed the orientation of the craft by more than $\pm 1/2^o$ would make the interpretation and combination of data for the planet as a whole rather complicated. However this accuracy would be more than adequate for photography where the pictures will be obtained in less than 1 second, and for the magnetic field measurements where the mechanics of the instruments prevent any better accuracy. The beam width of the transmission antenna is sufficiently broad ($5\text{-}1/2^o$) to easily accommodate a maximum variation of $\pm 1/2^o$ in orientation. The estimates of the weight of the altitude control system given later in the physical payload calculation, were derived from extrapolated Mariner R data. No explicit consideration has been given to the compensation required for correcting mis-orientation due to meteorite impact.

7.4 Elliptical Orbit Characteristics

The perijove of an orbiting spacecraft has been previously determined at 3 R_J. Several factors influence the periodicity of the orbit, and hence the apojove. It is desirable for obvious reasons to use as low a transmission rate as possible for the information stored during the observational part of the orbit, and hence a long period would be desirable. However, if one wishes to avoid blanking by the planet, or refraction of the transmitted beam by magneto-ionospheric effects only a fraction of the orbit is available for transmission. On the other hand, the longer the orbit time, the more stringent the reliability requirement becomes and the less short term observation of the environmental variations are possible.

We have chosen a period of 45 days, (apojove = 100 R_J) as being compatible with these considerations. Transmission of even multiple high resolution TV pictures can be achieved in a few days using bit rates of 50-100 b.p.s. (as is suggested below) while the 45 day period allows relatively frequent inspection of the planet. Such an orbit would require a terminal velocity correction of approximately 3 Km/sec, which quadruples the weight of the spacecraft required at Jupiter. A more circular orbit would impose even more severe penalties.

8. MISSION OUTLINES

Four classes of payload ranging from a minimum with a very simple experimental package to a more sophisticated spacecraft with comprehensive, high resolution experiments, a reactor electrical supply and a one kw telemetry system are outlined in Tables 8.1 through 8.3. The indicated accuracy of the gross weight has been derived from estimates of the error in the individual weight assignments.

These payloads have been considered in conjunction with one or more of the three hyperthetical launch vehicles in most cases, and a final high performance injection stage to evaluate the feasibility of several possible Jovian missions. As previously mentioned, no attempt has been made to define either optimum parking orbit or optimum staging.

Similarly, variations in choice of travel time of width of launch window, etc., can alter the mission parameters, particularly for an orbiter. Tables 8.4 through 8.9 give the parameters associated with six "feasible" missions chosen as illustrative.

A Jupiter mission using Vehicle 1 provides only one launch window in the next decade, and allows a minimal package to be sent past Jupiter. An added factor in considering such a mission is that it has been assumed that a high performance rocket in the 8500 lbs. class would be available (and indeed compatible) for stacking as the upper stage of Vehicle 1.

The missions using the Vehicles 2 and 3 appear to be much more rewarding. Considerable trade-off possibilities between payload, flight time, and mission complexity are presented by the use of these larger vehicles. A number of missions are compared in Table 8.10. Their potentialities seem sufficiently attractive to warrant further study.

Table 8.1

PAYLOAD A (450 lbs. nominal)

Item	Weight
Structure 10%	45
Attitude Control	60
Experimental Package (magnetometer + I. R. equipment and particle counters)	45
Telemetry (10 watts and 15 b. p. s.)	5
Magnetic and Meteorite Shield	30
C. C. and S.	15
Antenna (4' dish; i. e. 27 db)	30
Isotopic Power Supply 200 We	200
Mid-course Propulsion (20 m/sec)	30
Total Payload Weight	460 lbs $\pm$ 10 %

PAYLOAD A* (350 lbs.)

Optimization of data transmission rate - power supply rating and of battery power - isotopic power coupled with increased power efficiency of the transmitter or use of laser communications could lead to significant weight reduction. An optimistic value of 350 lbs. has been estimated as being "achievable".

Table 8. 2

PAYLOAD B (2000 lb. nominal)

Item	Weight
Structure 10 %	200
Attitude Control (for 500 days)	100
Experimental Package (all experiments including TV)	200
Telemetry (50 watts and 100 b. p. s.)	20
Magnetic and Meteorite Shield	150
C. C. and S.	100
Antenna (6 ft dish, i. e. 30 db)	100
Isotopic Power Supply 1 Kw	1000
Mid-Course Propulsion (20 m/sec)	75
Total Payload Weight	1945 lbs ± 10%

Table 8.3

PAYLOAD C (10,000 lbs. nominal)

Item	Weight
Structure 10%	1000
Attitude Control (for 500 days)	800
Experimental Package (all experiments including TV) *	200
Telemetry (1 Kw and 10^4 b.p.s.)	150
Magnetic and Meteorite Shield	1500
C. C. and S.	150
Antenna (15' or 38 db)	700
Power Supply (SNAP-8)	3500
Reactor Shield	1000
Mid-Course Propulsion 20 m/sec	500
Total Payload Weight	9500 $\pm$ 10%

* Additional experimental capability is an obvious possibility since the experiments constitute such a small fraction of the total payload weight. No explicit consideration has been given to added capability since it will not materially influence mission feasibility.

Table 8.4

MISSION I

Launch Vehicle	Vehicle 1
Parking Orbit	300 N. miles
Injection Stage Weight	7500-8500 lbs.
Payload A	450 lbs.
Flight Time	500 days
Mission Type	Flyby
Launch Constraint	One 30 day window in 1973, taking V_{HL} as 10.9 km/sec

Table 8.5

MISSION II

Launch Vehicle	Vehicle 2
Parking Orbit	300 N. miles
Injection Stage Weight	25,000 - 30,000 lbs.
Payload A	450 lbs.
Flight Time	400 days
Mission Type	Flyby
Launch Constraint*	30 day windows in 1972, 73, 74 and 75, taking V_{HL} as 13.4 km/sec

*By increasing the flight time to between 400 and 500 days, a 30 day launch window can be obtained annually.

Table 8.6

MISSION III

Launch Vehicle	Vehicle 2
Parking Orbit	300 N. miles
Injection Stage Weight	25,000 - 30,000 lbs.
Payload B	2,000 lbs.
Flight Time	500 days
Mission Type	Flyby
Launch Constraint	One 30 day window in 1973, taking V_{HL} as 10.9 km/sec

Table 8.7

MISSION IV

Launch Vehicle	Vehicle 3
Parking Orbit	300 N. Miles
Injection Stage Weight	180,000 - 220,000 lbs.
Terminal Propulsion	4,500 lbs.
Payload B	2,000 lbs.
Flight Time	500 days
Mission Type	Orbiter
Launch Constraint	Annual 30 day window, taking V_{HL} as 11.9 km/sec

Table 8.8

MISSION V

Launch Vehicle	Vehicle 3
Parking Orbit	300 N. miles
Injection Stage Weight	85,000 - 90,000 lbs.
Terminal Propulsion	3,200 lbs.
Payload B	2,000 lbs.
Flight Time	500 days
Mission Type	Orbiter
Launch Constraint	30 day launch window in 1972, 73, and 74, taking V_{HL} as 11 km/sec

Since the capability of Vehicle 3 is approximately 220,000 lbs. in a 300 N mile orbit, and here only 94,000 lbs. is needed, the possibility presents itself of partially burning the last stage of Vehicle 3, and re-igniting to provide the initial injection.

Table 8.9

MISSION VI

Launch Vehicle	Vehicle 3
Parking Orbit	300 N. miles
Injection Stage Weight	180,000 - 220,000 lbs.
Payload C	10,000 lbs.
Flight Time	500 days
Mission Type	Flyby
Launch Constraint	30 day windows annually from 1969 through 1976, taking V_{HL} as 11.5 km/sec

Table 8.10

COMPARISON OF MISSIONS TO JUPITER

Launch Vehicle	Mission Type	Payload (lbs.)	Ideal Velocity (ft/sec)	Flight Time (days)	Launch Years	Launch Window (days)	Remarks
Vehicle 1 plus solid propellant stage	Flyby	A*-350	53, 200	800	1969-75	20	
	Flyby	A*-350	53, 200	600	1969-75	25	
	Flyby	A*-350	53, 200	500	1972	10	
	Flyby	A-450	52, 000	800	1969-72	20	
	Flyby	A-450	52, 000	600	1971-72	15	
Vehicle 2 plus LH_2/Lox stage (30, 000 lbs.) plus solid propellant stage	Flyby	A*-350	59, 000	400	1973 and 74	20	
	Flyby	A-450	58, 000	400	1973 and 74	15	
	Flyby	B-2000	51, 000	1000	1969	25	
	Flyby	B-2000	51, 000	800	1969	10	
	Flyby	B-2000	51, 000	800	1971	15	
Vehicle 3	Flyby	B-2000	58, 500	400	1973 and 74	20	
	Flyby	B-2000	58, 500	500	Annual	30	Excess capability available for most opportunities.
	Orbiter	B-2000	56, 500	600	Annual	30	Allowance has been made for 2500 lbs. terminal propulsion.
Vehicle 3 plus LH_2/Lox stage (30, 000 lbs.)	Flyby	B-2000	63, 500	Approx. 350	1974	10	
	Orbiter	B-2000	57, 000	500	Annual	30	Allowance has been made for 7350 lbs. terminal propulsion.
	Orbiter	C-10, 000	52, 000	800	1969 and 71	20	Allowance has been made for 11, 600 lbs. terminal propulsion

9. CONCLUSIONS AND COMMENTS

The Jovian Mission Study started with a survey of the scientific objectives of investigating Jupiter with a space probe, and has been followed through with the configuration of the payload, the designation of launch vehicles and the launch windows available for a number of trajectories.

Scientifically, Jupiter is extremely interesting and a number of objectives have been established which are both valuable and suitable for determination from a space probe with a reasonable chance of success. The scientific payload has been built up using largely 'stage of the art' instrumentation, some of which has already been flight tested, and should therefore provide reliable data from the planet. However, the individual scientific experiments use but a small percentage of the total payload weight and it should be borne in mind that changes can be made relatively easily, if advances in the knowledge of Jupiter or the intervening space demand this.

The establishment of the scientific value of a Jovian mission has led to an evaluation of the many parameters influencing the overall feasibility of such a mission. Numerous assumptions have been made and the exercise of judgment and compromise throughout the study has resulted in the final mission proposals. The various areas where these assumptions are probably the most far reaching are summarized below.

(a) An early decision was required to establish which generation of launch vehicles would be most suitable for the Jovian mission and, in particular, whether nuclear-chemical rockets would be required. It has been shown that the vehicles suggested, with escape capabilities of 2,000-80,000 lbs. would be adequate, assuming that the final stage could be used to optimize the system after a parking orbit had been established.

(b) A number of detailed trajectories have been computed assuming that the spacecraft is under the influence of only one gravitating body at any one time, and for the transfer trajectory that the sun is the center of attraction. Impulsive acceleration has also been assumed and has resulted in the trajectories being simple conic sections.

(c) In assessing the value of a particular trajectory in terms of flight time, velocity increment and launch date, certain emphasis has been given to the launch windows which have been stipulated as a minimum of 30 days thus allowing for adequate checking of the system on the launch pad and for meteorological conditions at launch.

(d) The final stage has not been specified in any detail but requires a high performance propulsion system which will be compatible with the earlier stages.

(e) Mid-course guidance has only been considered to a zero order of sensitivity and has assumed the use of a storable fuel and a rocket with a restart capability. The major error assumed has been in the cut-off velocity and not in the direction of thrust.

(f) Mission boundaries have been established, but perhaps can be considered in more detail in any further study. The reliability of the spacecraft systems is simply assumed to be compatible with the duration of the mission. The hazards of a deep space flight to Jupiter have been considered but relatively little data is available for the region between earth and Jupiter. In particular, rather optimistic assumptions about meteor collisions have been used to estimate the protective shielding weight required and no consideration has been given to the cumulative deflection of the spacecraft due to these collisions, or to the power requirements for correcting such effects.

(g) The missions proposed include both flyby and orbiting probes and all should give a valuable amount of data on Jupiter. The mission using Vehicle 1 is, however, limited in its launch window and in its experimental payload. The more versatile vehicles 2 and 3 allow a range of compromises between the payload, flight time and mission complexity.

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